RECEIVED 11-6-67
MS-DV
V-SE00014772R C01

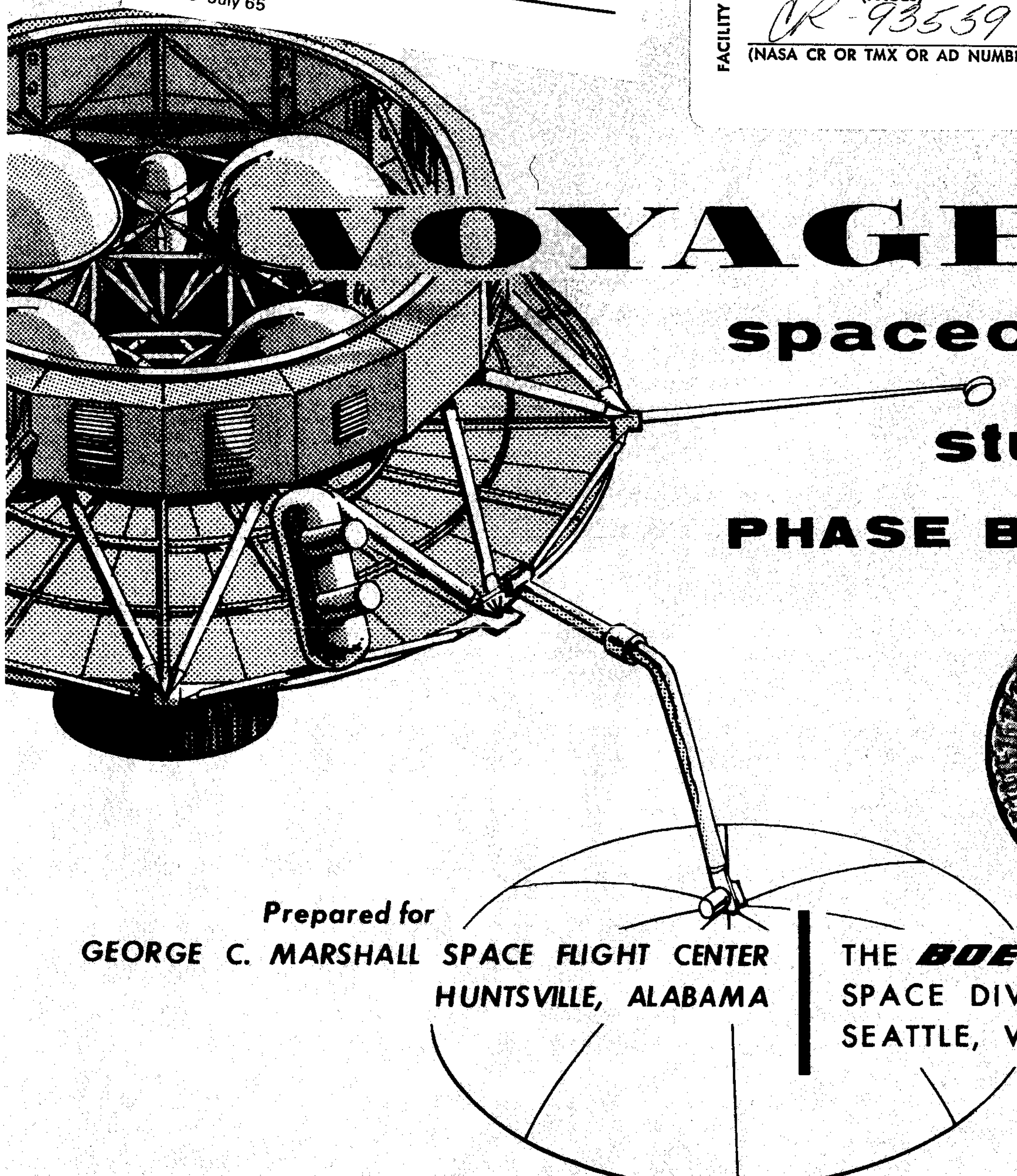
Volume III D2-115002-3

SPACECRAFT FUNCTIONAL DESCRIPTION

OCTOBER, 1967

GPO PRICE $ ______
CFSTI PRICE(S) $ ______
Hard copy (HC) 3.00
Microfiche (MF) .65
ff 653 July 65

FACILITY FORM 602
N68-19097 (ACCESSION NUMBER)
(THRU)
365 (PAGES)
1 (CODE)
CR-93559 (NASA CR OR TMX OR AD NUMBER)
31 (CATEGORY)



VOYAGER

spacecraft system studies

PHASE B, TASK D

Prepared for
GEORGE C. MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA

THE BOEING COMPANY
SPACE DIVISION
SEATTLE, WASHINGTON

VOYAGER

SPACECRAFT SYSTEM STUDY

•

FINAL TECHNICAL REPORT

PHASE B, TASK D

VOLUME III
SPACECRAFT FUNCTIONAL DESCRIPTION

D2-115002-3

OCTOBER 1967

Prepared For:

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GEORGE C. MARSHALL SPACE FLIGHT CENTER

HUNTSVILLE, ALABAMA

UNDER

CONTRACT NO. NAS8-22602

THE BOEING COMPANY • SPACE DIVISION • SEATTLE, WASHINGTON

FOREWORD

This series of documents summarizes the work performed under the George C. Marshall Space Flight Center contract, NAS 8-22602 entitled, "Voyager Spacecraft System, Phase B, Task D." The work was performed over the period June 16 through October 16, 1967.

The contracted work consisted of engineering studies leading to a definition of a Voyager Mars spacecraft system capable of performing the 1973 mission. To ensure flexibility of design, additional analyses were conducted to determine the adaptability of the 1973 spacecraft to perform the 1975-1977-1979 Mars missions. The 1973 flight spacecraft definition was used to identify the operational support equipment including mission-dependent equipment requirements and the software necessary to satisfactorily conduct the 1973 mission operations. Logistics considerations were identified for the 1973 system from point of manufacture through launch operations.

The contract also required the completion of five selected engineering tasks that were designed to highlight key areas and lead to specific conclusions and recommendations.

The detailed results of the contracted work is contained in the following reports:

- <u>Summary Report</u> — Volume I, D2-115002-1
- <u>Mission/System Requirements and Analyses</u> — Volume II, D2-115002-2
- <u>Spacecraft Functional Description</u> — Volume III, D2-115002-3
- <u>Selected Engineering Tasks</u> — Volume IV, D2-115002-4

CONTENTS

1.0

INTRODUCTION AND SUMMARY

1.1 SYSTEM LEVEL FUNCTIONAL DESCRIPTION

1.0 INTRODUCTION AND SUMMARY

This document is the third volume of the Phase B, Task D Final Technical Report on Voyager spacecraft system studies performed by The Boeing Company. These studies were conducted for the George C. Marshall Space Flight Center under NASA Contract No. NAS8-22602.

The volume defines a spacecraft that is capable of performing the 1973 Voyager Mars mission and that is adaptable to perform the 1975, 1977, and 1979 Mars missions with minimum hardware changes.

Background and Objectives

Voyager spacecraft system definition phase studies have been conducted since early 1965. These studies have included concept refinements, assessment of preliminary requirements, and system analysis and tradeoffs. These studies resulted in the definition and functional description of a spacecraft system capable of performing the 1971 mission.

During Task D of the definition phase, the flight spacecraft was revised to reflect (1) a change in the first Mars Voyager launch opportunity from 1971 to 1973, and (2) a need to develop a spacecraft system that can be adapted to subsequent Mars Voyager missions in 1975, 1977, and 1979 with minimum modifications.

Mission/system requirement for the 1973, 1975, 1977, and 1979 Mars missions were prepared (see Volume II, D2-115002-2). Standard trajectories and associated parameters, guidance and navigation maneuver errors, and weight, reliability, and power requirements were established.

The spacecraft definition for the 1971 Mars mission, prepared in Phase B, was reviewed in light of these new requirements. Trade studies were performed where necessary, and a configuration selected (see Figure 1.1.7-1) that meets the 1973 mission requirements and that is easily adaptable to the 1975, 1977, and 1979 missions.

Summary

System Level Functional Description - Standard trajectories and associated parameters necessary for design of the 1973 mission spacecraft and for consideration of the 1975, 1977, and 1979 opportunities are included. Guidance and navigation maneuver errors are defined. A nominal flight sequence has been developed. Spacecraft interfaces with the launch vehicle, capsule, and science system are defined. Based on the above, and on many trade studies, a spacecraft configuration is defined. System level considerations such as weight, reliability, electrical power, particulate contamination, and planetary quarantine are also included.

Subsystem Functional Description - Each of the spacecraft subsystems has been described in terms of design constraints and requirements, functions, performance and physical characteristics, interfaces, and reliability. Summaries of trade studies performed during Task D for subsystem selection are included. New technology and development, if required, are defined and scheduled for each subsystem. The growth potential of each subsystem is discussed.

Spacecraft Changes Required for 1975, 1977 and 1979 Adaptability - The conceptual changes required are presented for the spacecraft configuration and each of the hardware subsystems. The changes required result from consideration of the following:

1) Effect of launch year on transit time, trajectories, etc.

2) Effects of science payload evolution discussed in D2-115002-4.

3) Effects of photoimaging considerations discussed in D2-115002-4.

1.1 SYSTEM LEVEL FUNCTIONAL DESCRIPTION

This section contains the system-level functional description of the preferred spacecraft design. Standard trajectories and associated parameters necessary for design of the 1973 mission spacecraft and for consideration of the 1975, 1977, and 1979 opportunities are included. Guidance and navigation maneuver errors are defined. A nominal flight sequence has been developed. Spacecraft interfaces with the launch vehicle, capsule, and science system are defined. Based on the above and on many trade studies, a spacecraft configuration is defined. System-level considerations such as weight, reliability, electrical power, particulate contamination, and planetary quarantine are also included.

1.1.1 Standard Trajectories

1.1.1.1 Purpose

The purpose of this section is to describe those standard trajectories, and associated parameters, that are necessary for the design of the spacecraft and its associated mission-dependent equipment. Criteria and methods used to select and calculate these trajectories are discussed in Section 3.1 of D2-115002-2.

1.1.1.2 Scope

The scope of this section is indicated in Table 1.1.1.-1 which relates paragraphs of this section to functional considerations fundamental to the design of the Voyager system.

1.1.1.3 Approach

In most instances, a single design trajectory will be shown. In cases where this is inadequate for design purposes, an envelope will be shown or a worst-case instance will be taken.

1.1.1.4 Launch Trajectory- 1973

Launch takes place from ETR on August 25, 1973. A nominal launch vehicle trajectory profile is shown in Figure 1.1.1.-1. Launch times, launch azimuths, and coast times are shown in Figure 1.1.1.-2.

Table 1.1.1-1: STANDARD TRAJECTORIES — Scope

VOYAGER SYSTEM, FUNCTIONAL CONSIDERATIONS	PERTINENT PARAGRAPH OF STANDARD TRAJECTORIES SECTION									
	Geocentric Position & Velocity	Topocentric Data For DSN Stations	Heliocentric Position	Cone & Clock Angles From S/C to Earth, Mars, Canopus, Capsule	Occultations of Sun, Earth, Canopus	ΔV Midcourse, Insertion, Trims	Guidance & Navigation Maneuver Errors	Spacecraft Capsule Communication	Orbit Photographic Coverage	Trajectory Characteristics 1975 - 1979
SPACECRAFT BASELINE DESIGN	1.1.1.4	1.1.1.5	1.1.1.6	1.1.1.10	1.1.1.10	1.1.1.6 1.1.1.7	1.1.1.11	1.1.1.8 1.1.1.9	1.1.1.10	1.1.1.12
SPACECRAFT HARDWARE SUBSYSTEM DESIGNS										
Power	1.1.1.4		1.1.1.6		1.1.1.10					1.1.1.12
Guidance and Control			1.1.1.6	1.1.1.10	1.1.1.10	1.1.1.6 1.1.1.7	1.1.1.11			1.1.1.12
Telecommunications	1.1.1.4	1.1.1.5	1.1.1.6	1.1.1.10	1.1.1.10			1.1.1.8 1.1.1.9	1.1.1.10	1.1.1.12
Propulsion						1.1.1.6 1.1.1.7	1.1.1.11			
Structures and Mechanisms				1.1.1.10	1.1.1.10	1.1.1.6 1.1.1.7				1.1.1.12
Temperature Control	1.1.1.4		1.1.1.6		1.1.1.10					1.1.1.12
Science			1.1.1.6		1.1.1.10			1.1.1.8 1.1.1.9	1.1.1.10	
OSE & MDE REQUIREMENTS	1.1.1.4	1.1.1.5	1.1.1.6		1.1.1.10					1.1.1.12
VOYAGER PROGRAM TEST FLIGHT	1.1.1.4	1.1.1.5	1.1.1.6							1.1.1.12

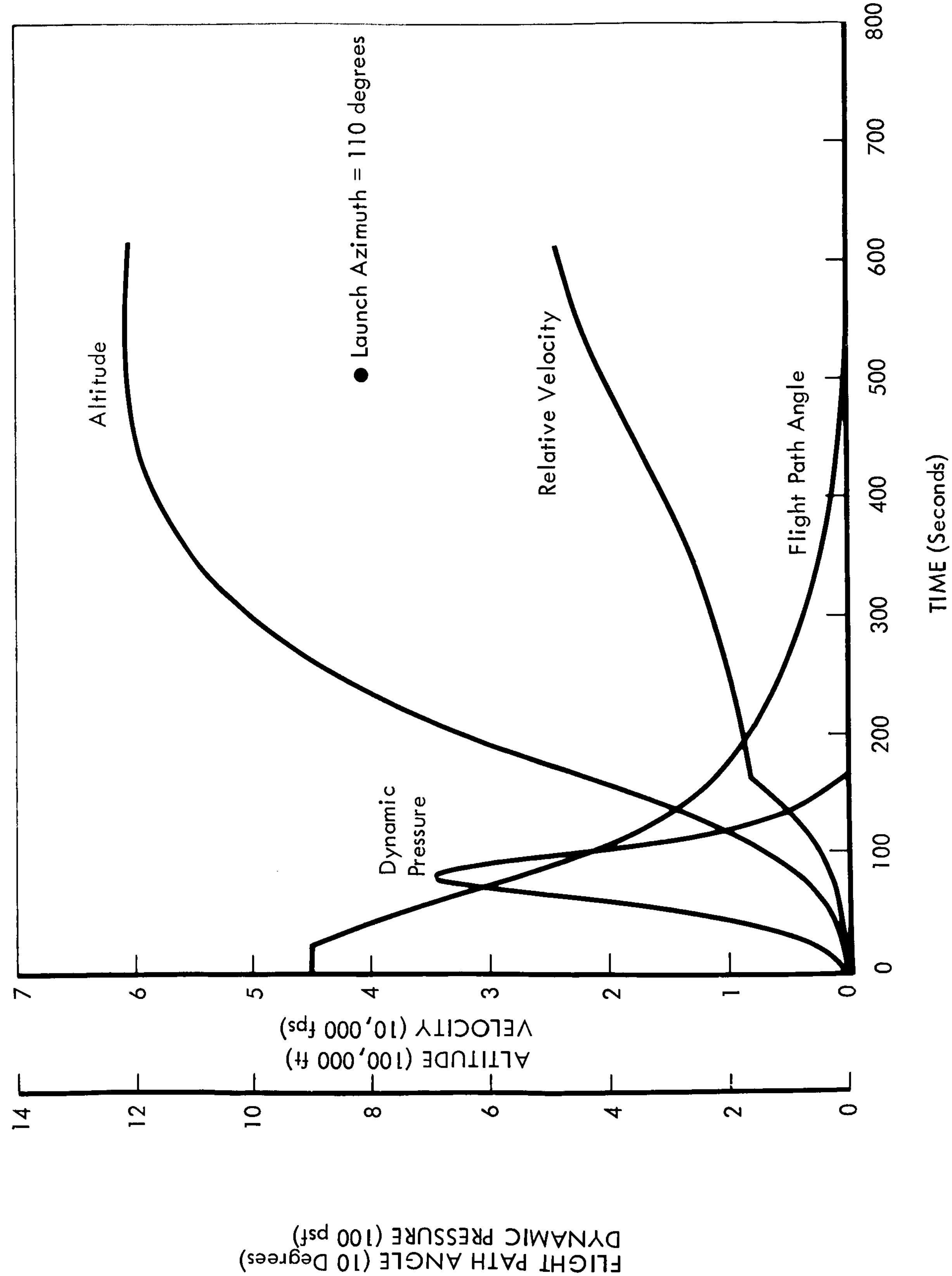


Figure 1.1.1-1: SATURN TRAJECTORY

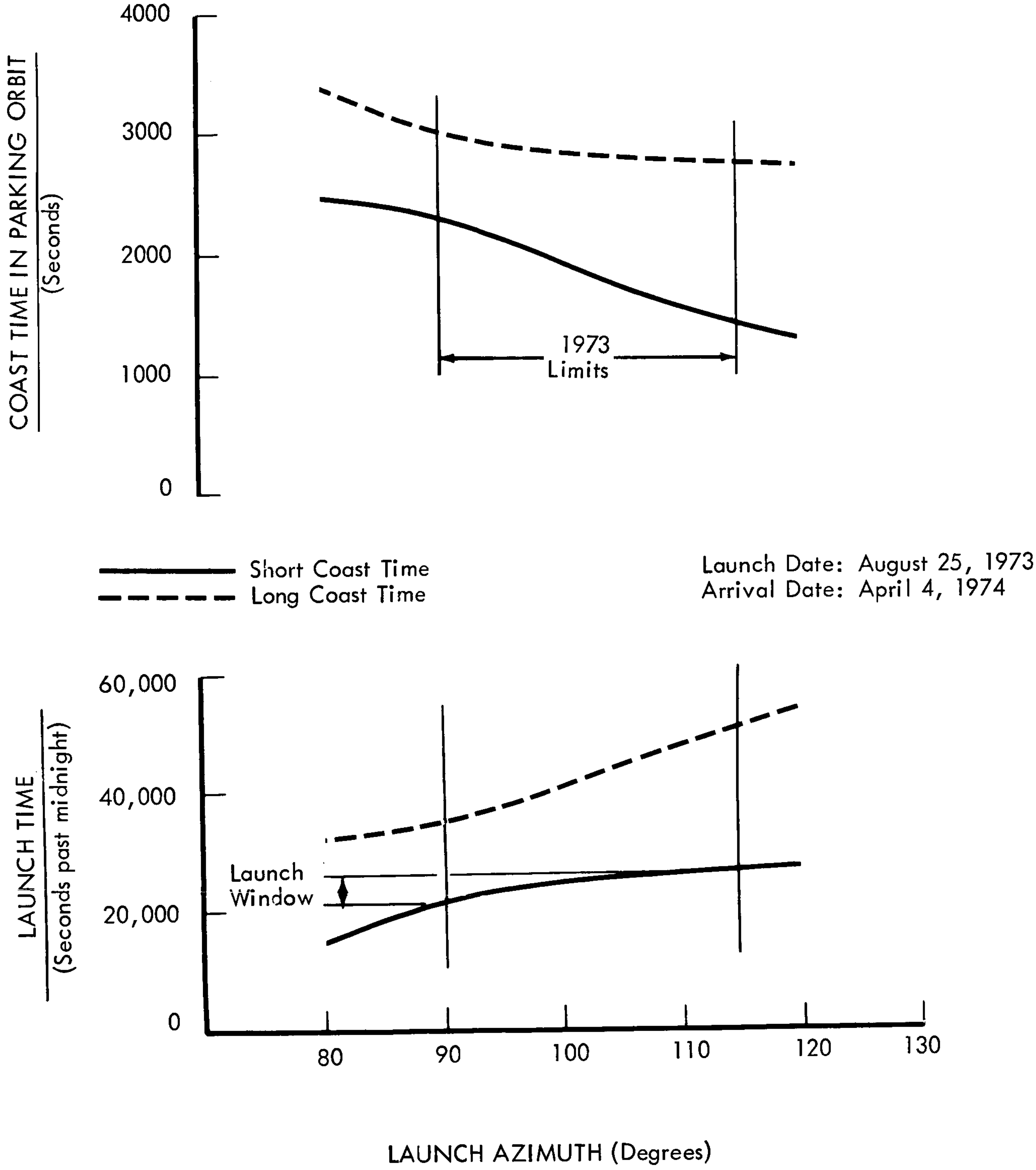


Figure 1.1.1-2: LAUNCH WINDOW & COAST TIME IN PARKING ORBIT

1.1.1.5 Near Earth Trajectories, 1973

Near Earth trajectories are shown in Figure 1.1.1-3. These cover typical launch azimuths for the short coast time in parking orbit. Tracking data for the Woomera DSN site are shown in Figure 1.1.1-4. Similar data have been calculated for the Madrid, Goldstone, and Johannesburg DSN stations.

1.1.1.6 Trans-Mars Trajectory, 1973

Figure 1.1.1-5 shows the trans-Mars trajectory as viewed from the north ecliptic pole. At launch, the aim point is offset from the planet, as shown in Figure 1.1.1-6, to keep the probability of contaminating the planet below 2×10^{-6}.

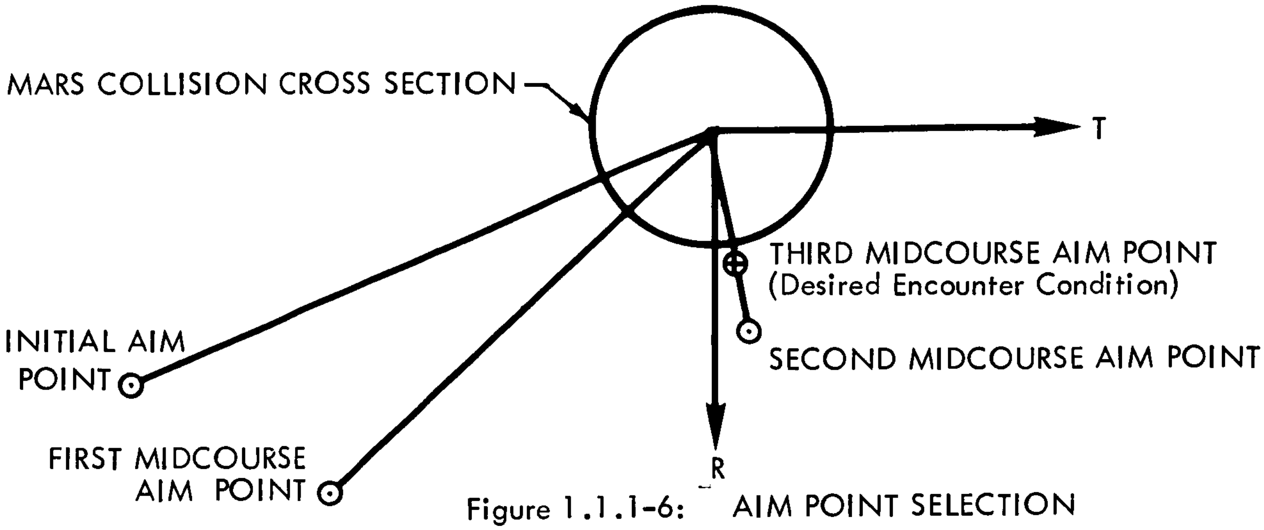


Figure 1.1.1-6: AIM POINT SELECTION

At the first midcourse correction, three functions are performed simultaneously: first, the arrival time of the two planetary vehicles is separated by 8 days; second, the insertion errors in aim point position and arrival time are nulled, and finally, the aiming point is shifted to a position appropriate for the second midcourse maneuver. Since the first midcourse maneuver will involve some errors in control, a second midcourse maneuver is planned. Similarly, a third midcourse maneuver may be executed. The total ΔV allocated for the midcourse maneuvers (including arrival time biasing) is 210 m/sec. Finally, the two planetary vehicles will arrive at Mars on March 31, 1974, and April 8, 1974.

The Sun-spacecraft distance, spacecraft-Earth communication distance, and the position of Earth with respect to the spacecraft during transit are discussed in Section 1.1.1.12.

1.1.1.7 Orbit Insertion and Trim

The design orbit about Mars is described in Table 1.1.1-2. Following insertion of a planetary vehicle into orbit, an orbit determination is made. Then, if required, an orbit trim is performed to adjust the existing orbit to conform with the mission requirements. Orbit trim will consist of two different maneuvers: first, periapsis radius, apoapsis radius, and argument of periapsis are corrected; this maneuver takes place at a true anomaly lying between 110 and 130 degrees. Then, following another orbit determination, the second trim maneuver will correct radius of periapsis,

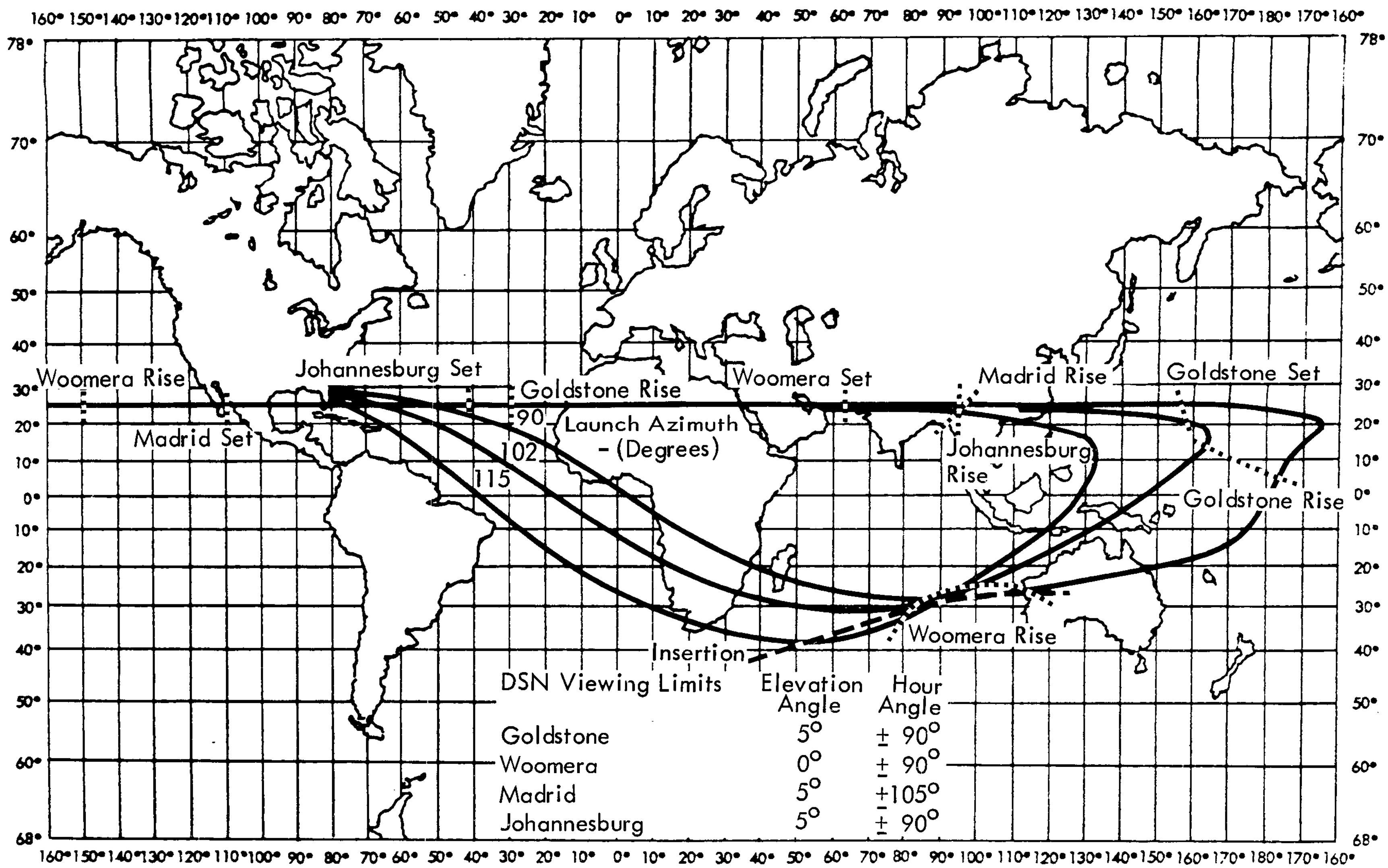


Figure 1.1.1-3: GROUND TRACKS FOR TRAJECTORIES LAUNCHED ON AUGUST 25, 1973 AND ARRIVING ON APRIL 4, 1974

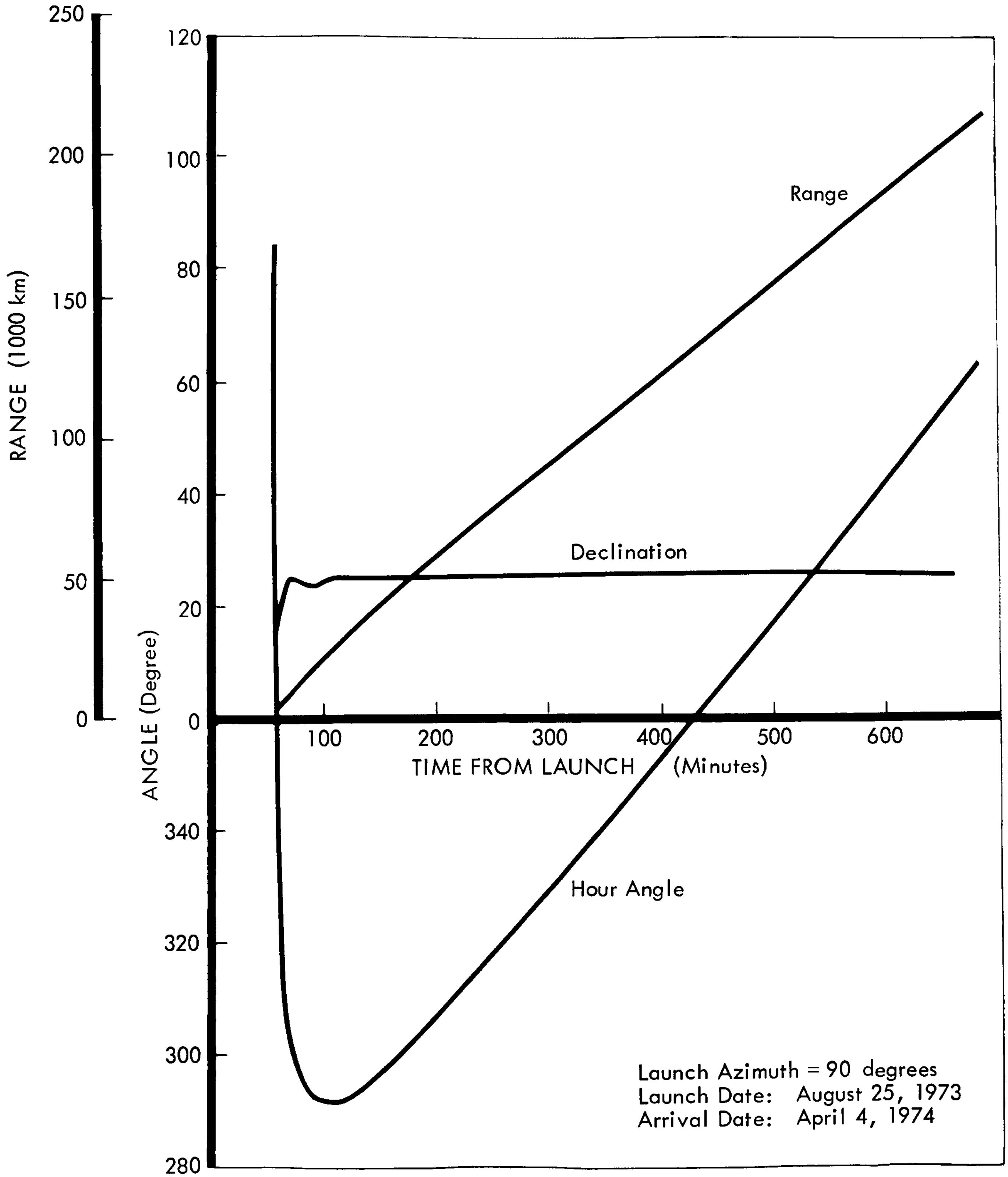


Figure 1.1.1-4: TRACKING DATA, WOOMERA STATION

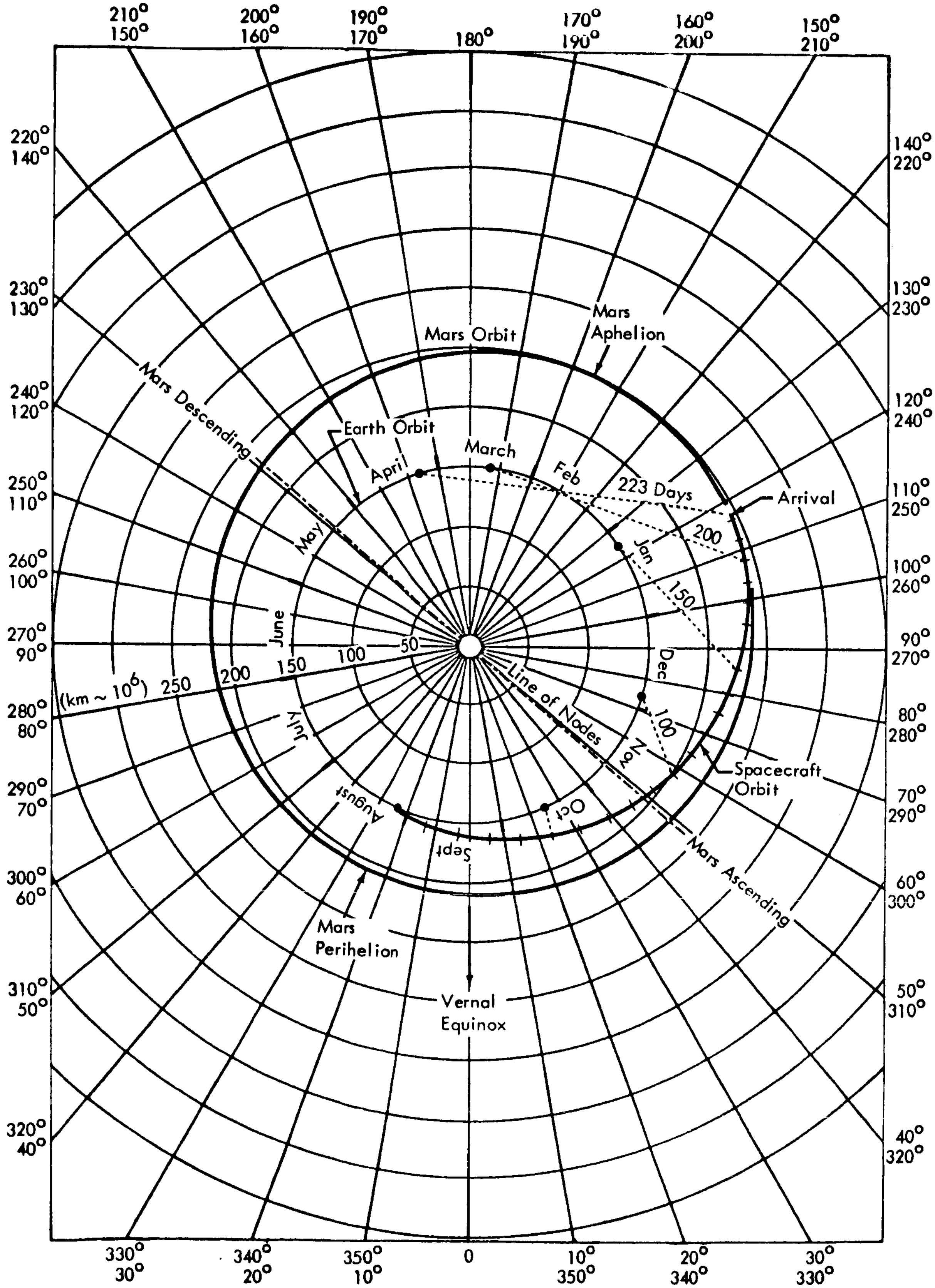


Figure 1.1.1-5: TYPICAL 1973 TYPE I INTERPLANETARY TRAJECTORY

Table 1.1.1-2: DESIGN ORBIT CHARACTERISTICS

Periapsis Altitude	500 Kilometers
Period	12.4 Hours
Apoapsis Altitude	18,590 Kilometers
Inclination to Martian Equator	+45 Degrees
Time of First Periapsis Passage	April 4, 1974 (0:00 Hours GMT)
Longitude of Ascending Node	289.4 Degrees
Argument of Periapsis	167.5 Degrees
Launch Date	August 25, 1973
Initial Illumination Angle at Periapsis	15 Degrees
Apsidal Rotation Required at Insertion	33.6 Degrees
ΔV Required for Insertion	1163 Meters/Second

radius of apoapsis, and longitude of node; this maneuver's true anomaly is situated between 210 and 240 degrees. The total ΔV allocated for trim is 150 m/sec.

1.1.1.8 Spacecraft-To-Capsule Communication Kinematics

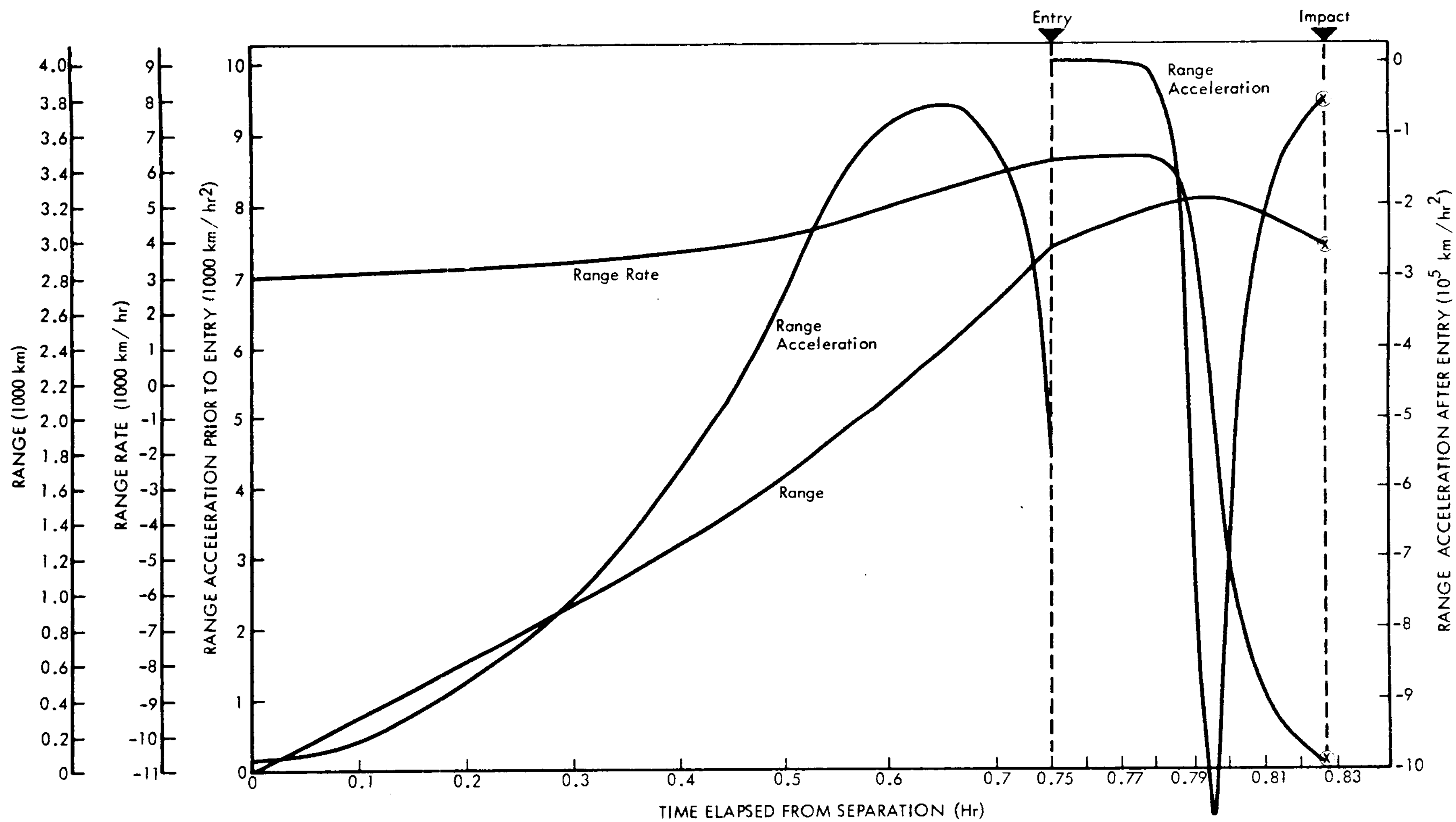
Following orbit trim, the capsule will be separated at a true anomaly between 220 and 250 degrees, for impact at a point corresponding to a true anomaly of 350 degrees on the spacecraft's orbit. Parameters critical to the design of the spacecraft capsule are shown in Figure 1.1.1-7. These data were calculated for an earlier version of the design orbit. The present orbit does not differ sufficiently to warrant recalculation of the data.

1.1.1.9 Spacecraft-To-Surface Laboratory Communication Kinematics

Figures 1.1.1-8 and 1.1.1-9 show the kinematic relationships between the spacecraft and the landed laboratory.

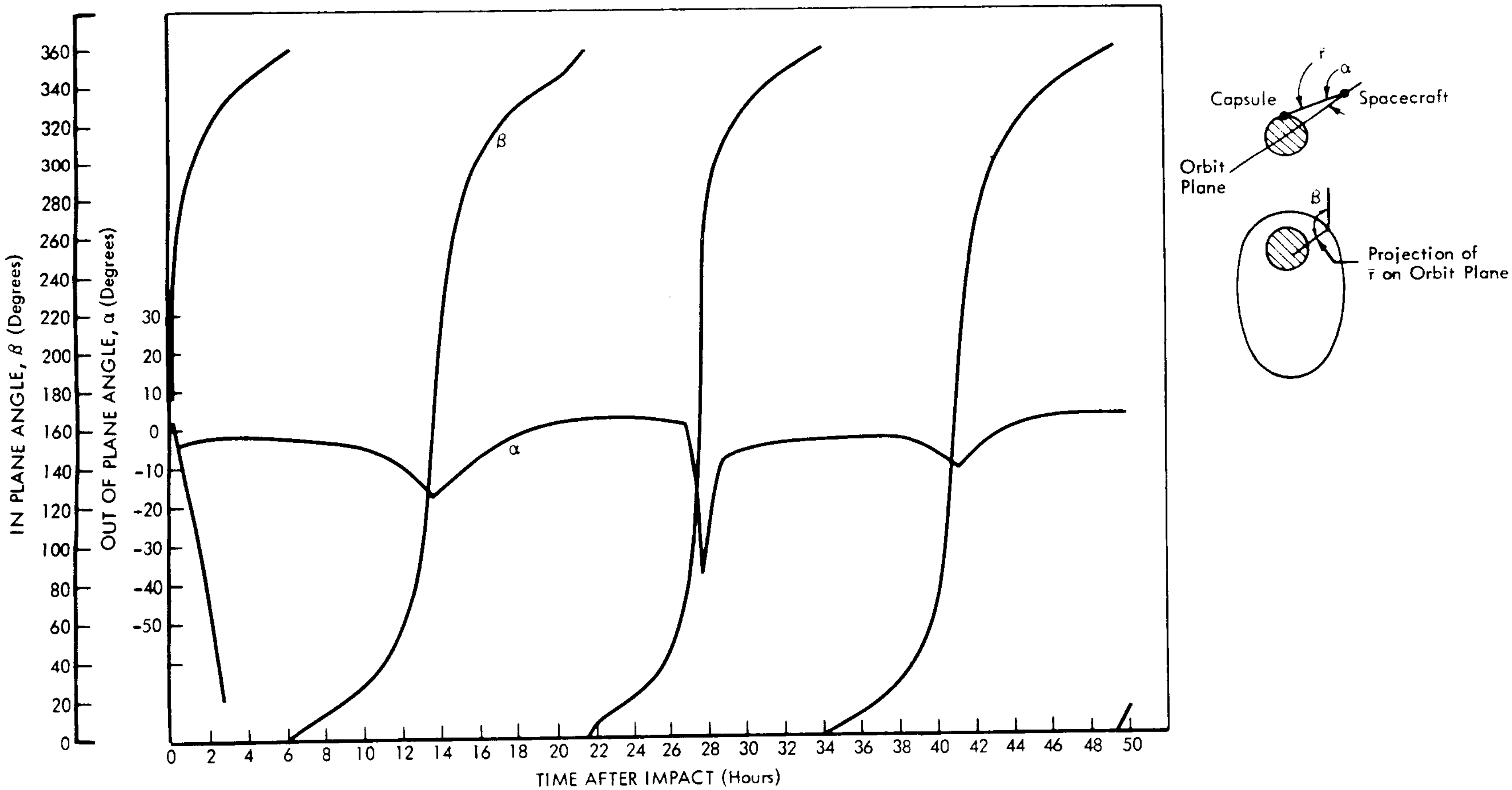
1.1.1.10 Orbit About Mars

To take science data from orbit, the spacecraft must be able to observe the entire disk of the planet. This requires moving the science platform throughout the range of cone and clock angles indicated in Figure 1.1.1-10. Figure 1.1.1-11 shows the time variation of the cone and clock angles of the center of Mars. The orbit for which Figures 1.1.1-10 and 1.1.1-11 were calculated was deliberately selected as being more critical from the cone and clock angle standpoint than the nominal design orbit.



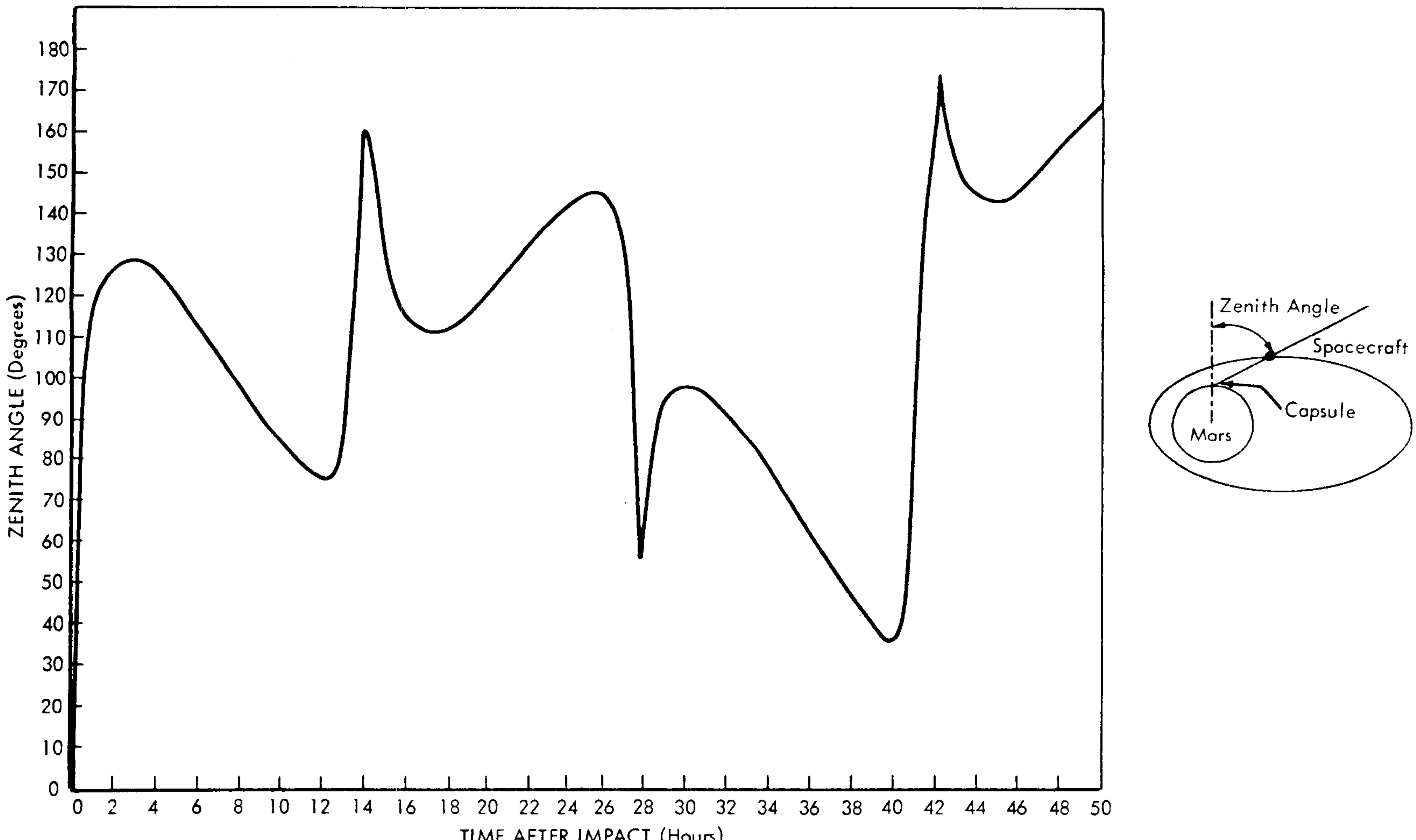
Notes:
- VM-8 Atmosphere (5 mb)
- M/C_DA of Capsule = 0.6 slug/ft^2
- Launch Date = August 15, 1973
- Arrival Date = April 4, 1974
- Periapsis Altitude = 1000 km
- Orbit Period = 14 hours
- Inclination = -30 degrees
- Illumination = 15 degrees from Evening Terminator
- Separation at True Anomaly = 250 degrees
- Impact at True Anomaly = 350 degrees

Figure 1.1.1-7: MOTION OF CAPSULE WITH RESPECT TO SPACECRAFT



Notes:
- Launch Date: August 15, 1973
- Arrival Date: April 4, 1974
- Periapsis Altitude = 1000 km
- Orbit Period = 14 hours
- Inclination = -30 degrees
- Illumination = 15 degrees from Evening Terminator

Figure 1.1.1- 8: SPACECRAFT-CAPSULE VIEW ANGLES



Notes:
- Launch Date: August 15, 1973
- Arrival Date: April 4, 1974
- Periapsis Altitude = 1000 km
- Orbit Period = 14 hours
- Inclination = -30 degrees
- Illumination = 15 degrees from Evening Terminator

Figure 1.1.1-9: SPACECRAFT ZENITH ANGLE — Relative to Capsule

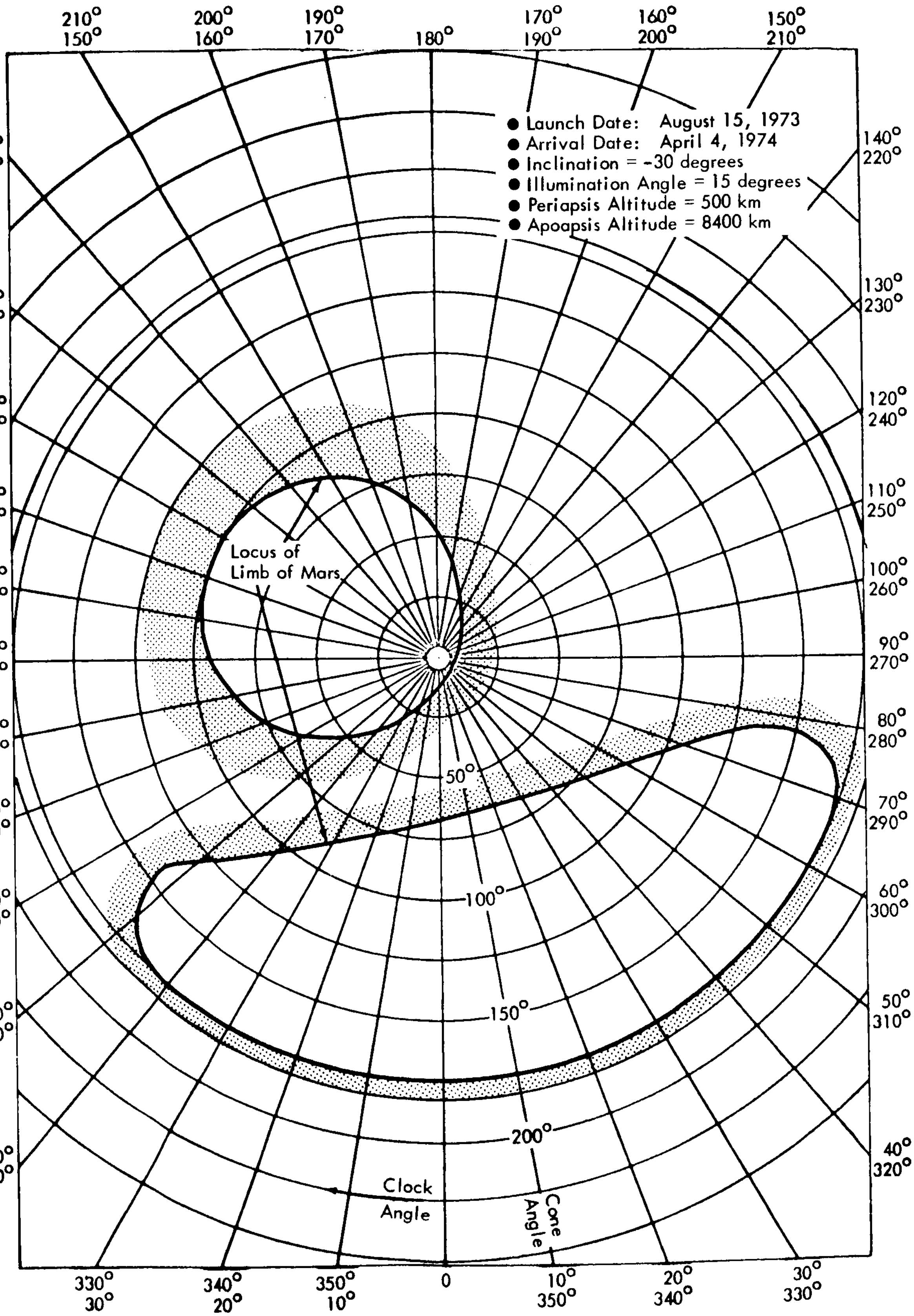


Figure 1.1.1-10: CONE AND CLOCK ANGLES OF LIMB OF MARS

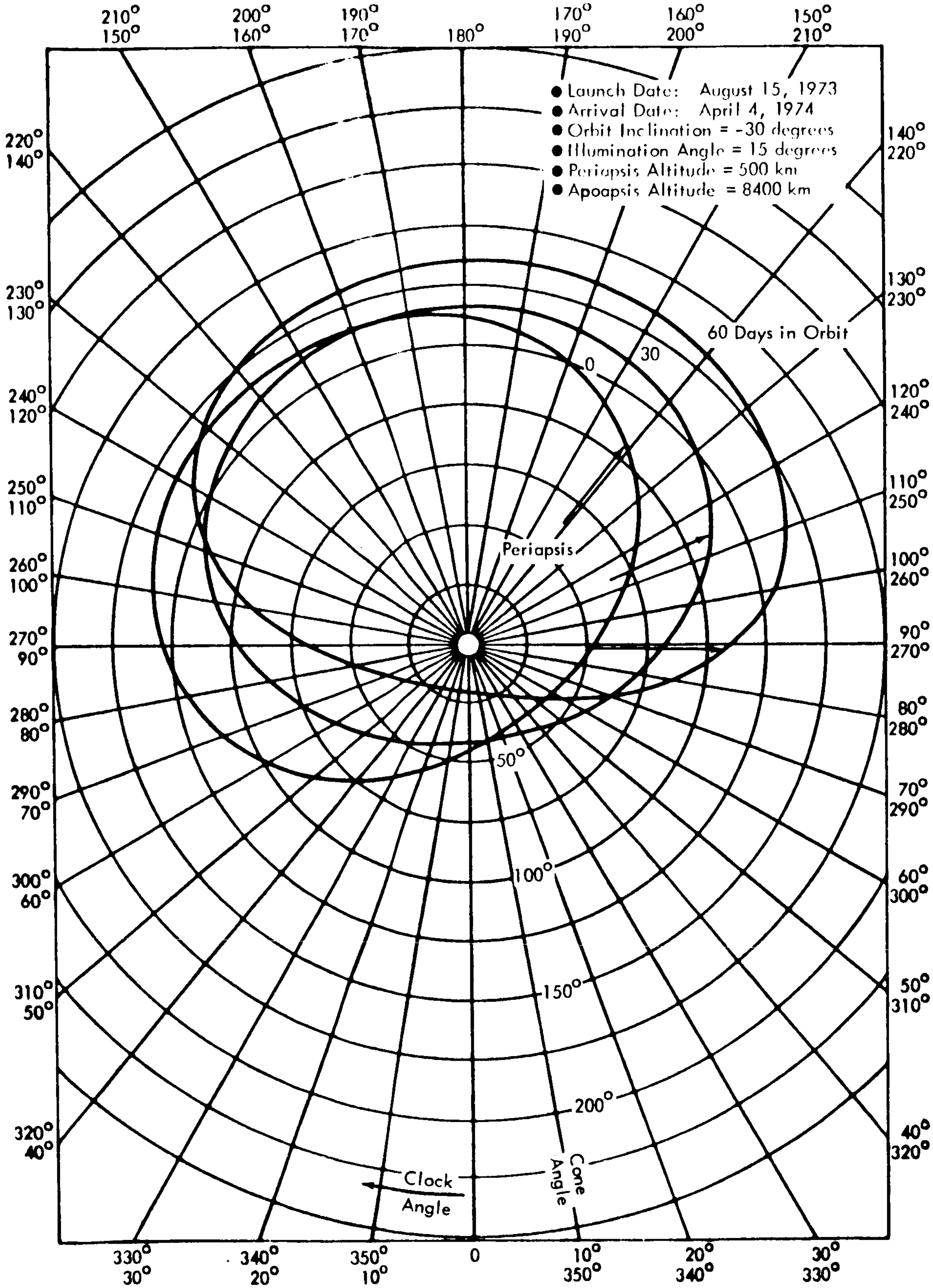


Figure 1.1.1-11: CONE AND CLOCK ANGLE OF CENTER OF MARS

The ground track of the design orbit is shown in Figure 1.1.1-12. The ability of this orbit to observe the wave of darkening is illustrated in Figure 1.1.1-13. The band of latitudes with favorable illumination is shown in Figure 1.1.1-14 as a function of time in orbit.

Figures 1.1.1-15, -16 and -17 describe situations in which Mars occults Sun, Canopus, and Earth, respectively. Solar occultations affect the power subsystems and temperature control subsystem designs. Solar and Canopus occultations affect the spacecraft attitude control subsystem, and Earth occultation affects the telecommunications subsystem design, and also the design of the atmospheric occultation experiment. In the case of Canopus, there is no actual physical occultation during the 180-day orbit duration. Consequently, only periods of interference are defined. The latitude at which the Earth rises and sets with respect to the spacecraft is shown in Figure 1.1.1-18.

Solar, Earth, and Canopus occultations are said to occur whenever the spacecraft-body line of sight lies within 0 degrees of the near limb of Mars. Canopus interference occurs whenever any lighted portion of Mars intrudes within the pyramidal confines of the Canopus tracker shield. For this study, the shield limits were set at ± 30 degrees in spacecraft-fixed cone angle and ± 20 degrees in spacecraft fixed clock angle.

1.1.1.11 Requirements on Control Accuracy

Control error set C, the selected spacecraft control error set per the trade study in Section 3.1.5, D2-115002-2, requires the following performance:

Error	3σ Value
Pointing, Total	42 mrad
ΔV, Proportional	1.0% Midcourse, 0.15% Insertion
ΔV, Resolution*	0.02 m/sec

*Excluding engine tailoff uncertainties which were considered separately.

1.1.1.12 Consideration of the Opportunities in 1975, 1977 and 1979

Type I trajectories have been computed for launch/arrival date combinations that cover the entire operational envelope for 1973, 1975, 1977, and 1979. The envelopes of the cone and clock angles of Earth as seen from the planetary vehicle are shown in Figures 1.1.1-19 through 1.1.1-22. Communication distances from the planetary vehicle to Earth have been treated in similar fashion in Figures 1.1.1-23 through 1.1.1-26. The corresponding Sun-spacecraft distances are illustrated in Figures 1.1.1-27 through 1.1.1-30. These data are significant to the design of the telecommunications system and in the placement of the high gain antenna. They are also necessary to the design of the power and temperature control subsystems.

The operations described in the preceding paragraph have been repeated for Type II trajectories during the 1975 and 1977 opportunities. The results are shown in Figures 1.1.1-31 through 1.1.1-36.

- Launch Date: August 25, 1973
- Arrival Date: April 4, 1974
- Periapsis Altitude = 500 km
- Orbit Period = 12.4 hours
- Inclination = +45 degrees

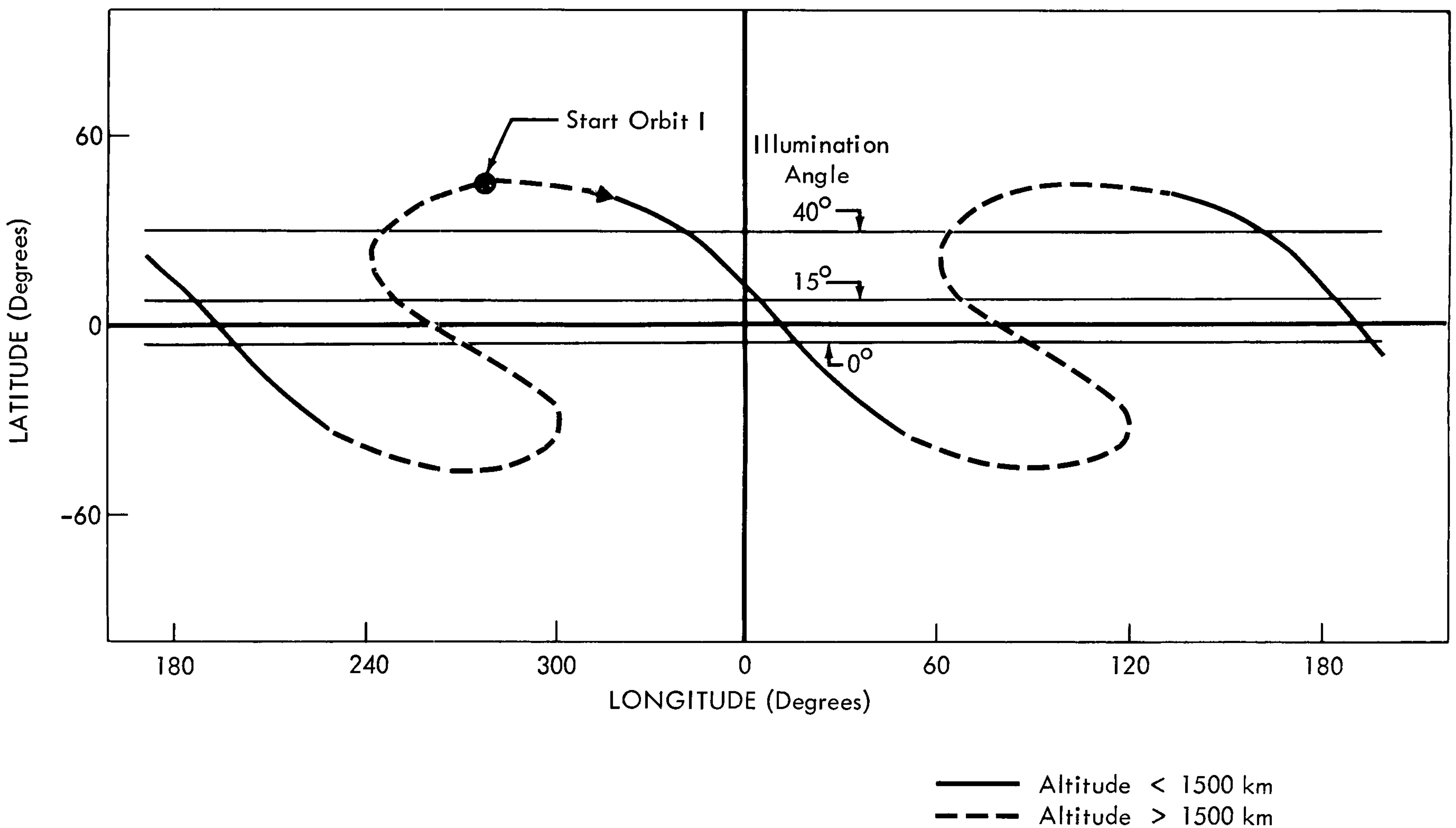


Figure 1.1.1-12: DESIGN ORBIT ILLUMINATION BAND — First Day

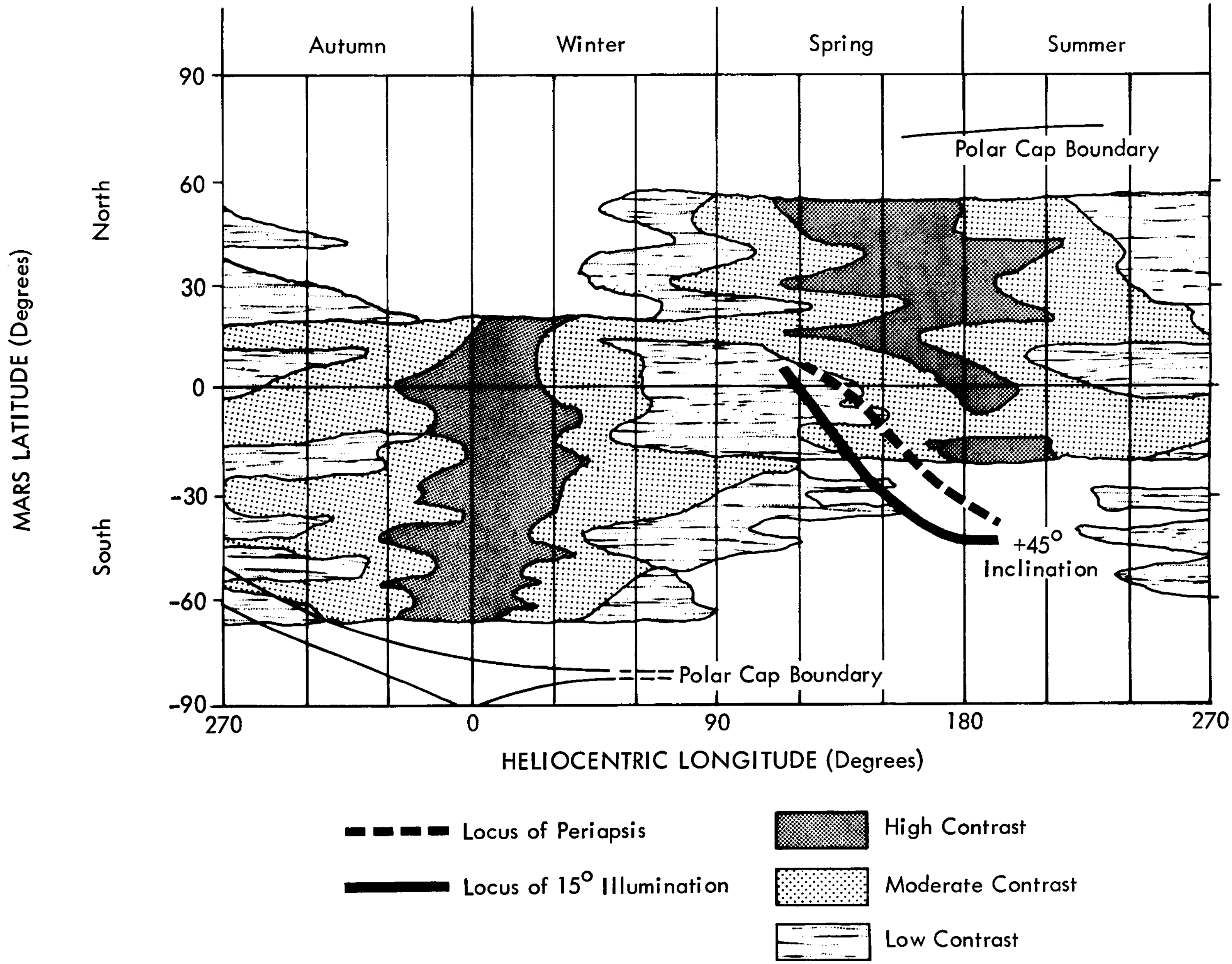


Figure 1.1.1-13: DESIGN ORBIT OBSERVATION OF WAVE OF DARKENING

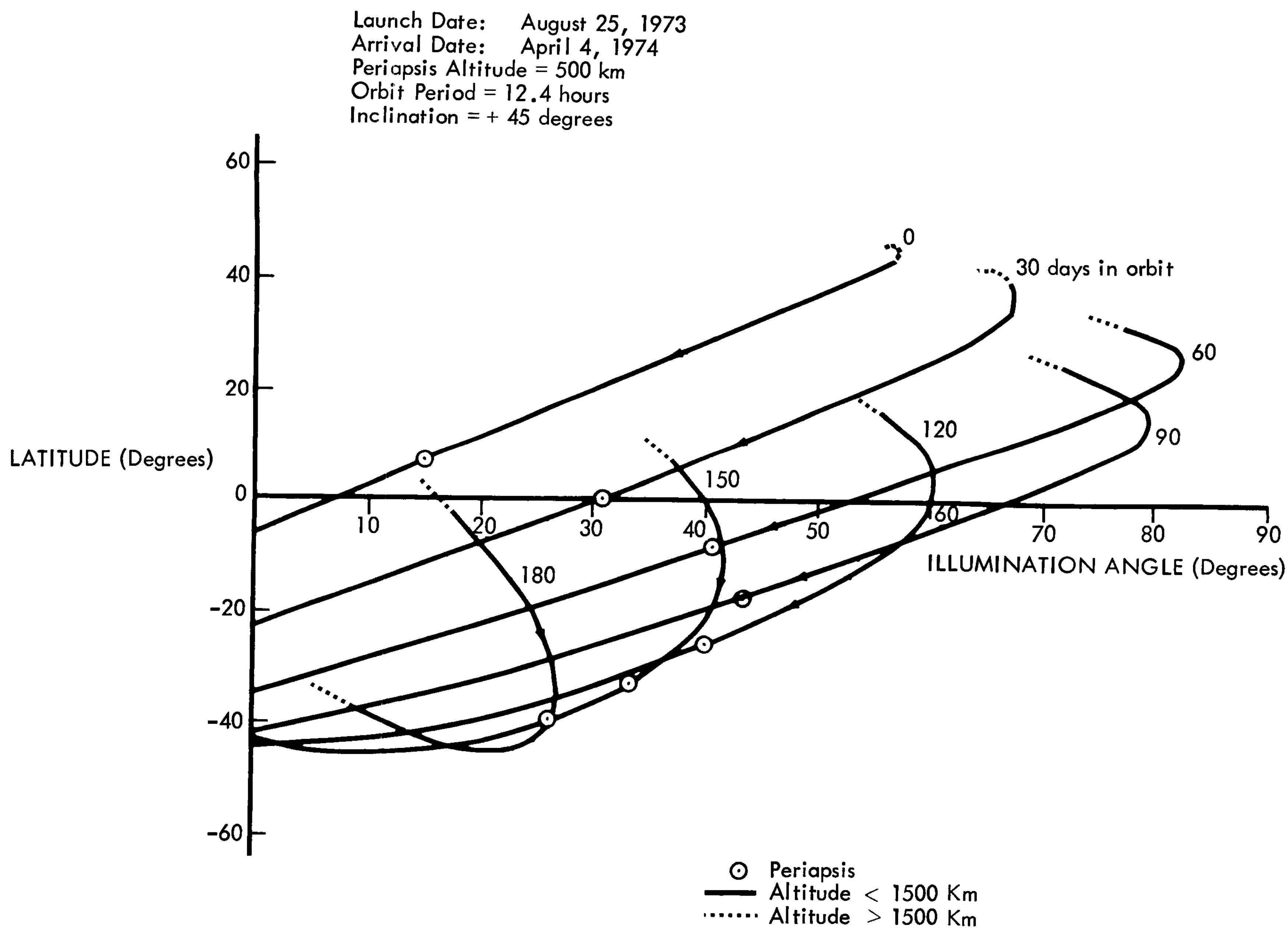


Figure 1.1.1-14: DESIGN ORBIT PHOTOGRAPHIC COVERAGE

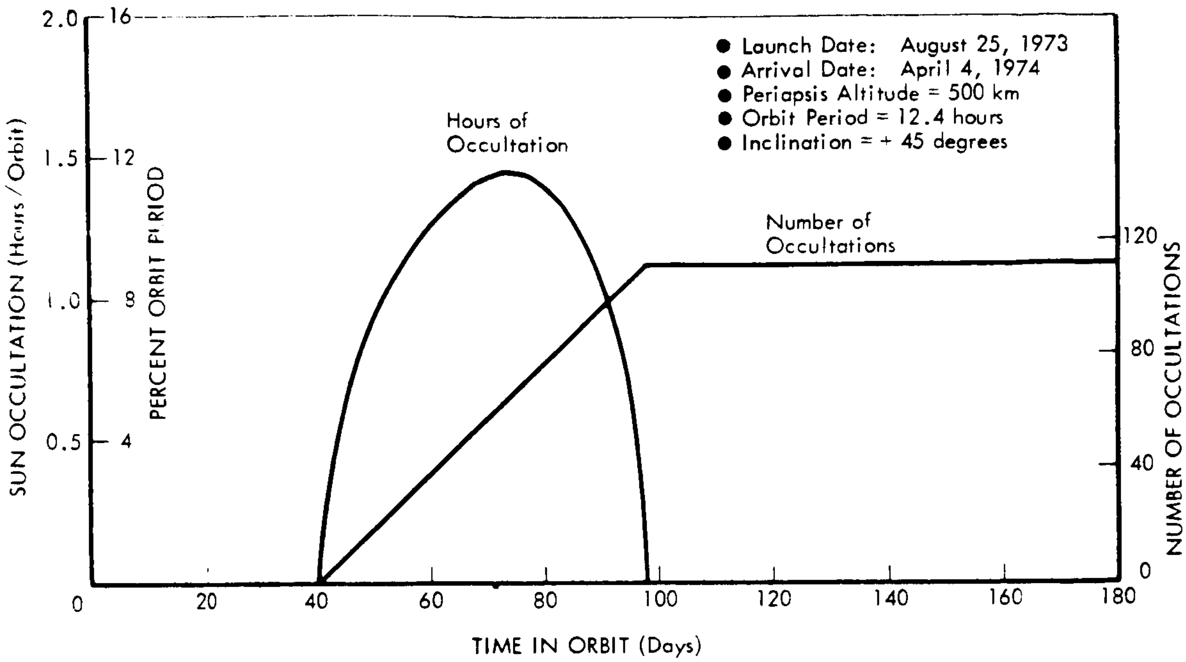


Figure 1.1.1-15: DESIGN ORBIT SUN OCCULTATION

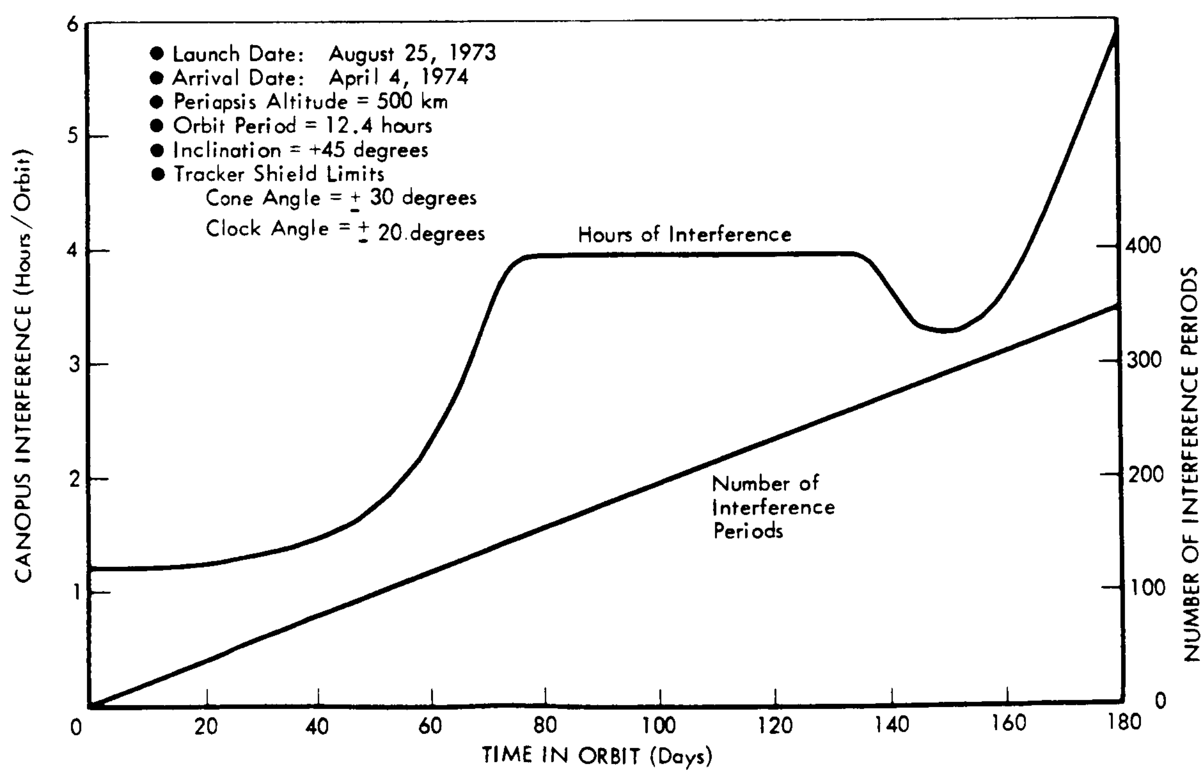


Figure 1.1.1-16: DESIGN ORBIT CANOPUS INTERFERENCE

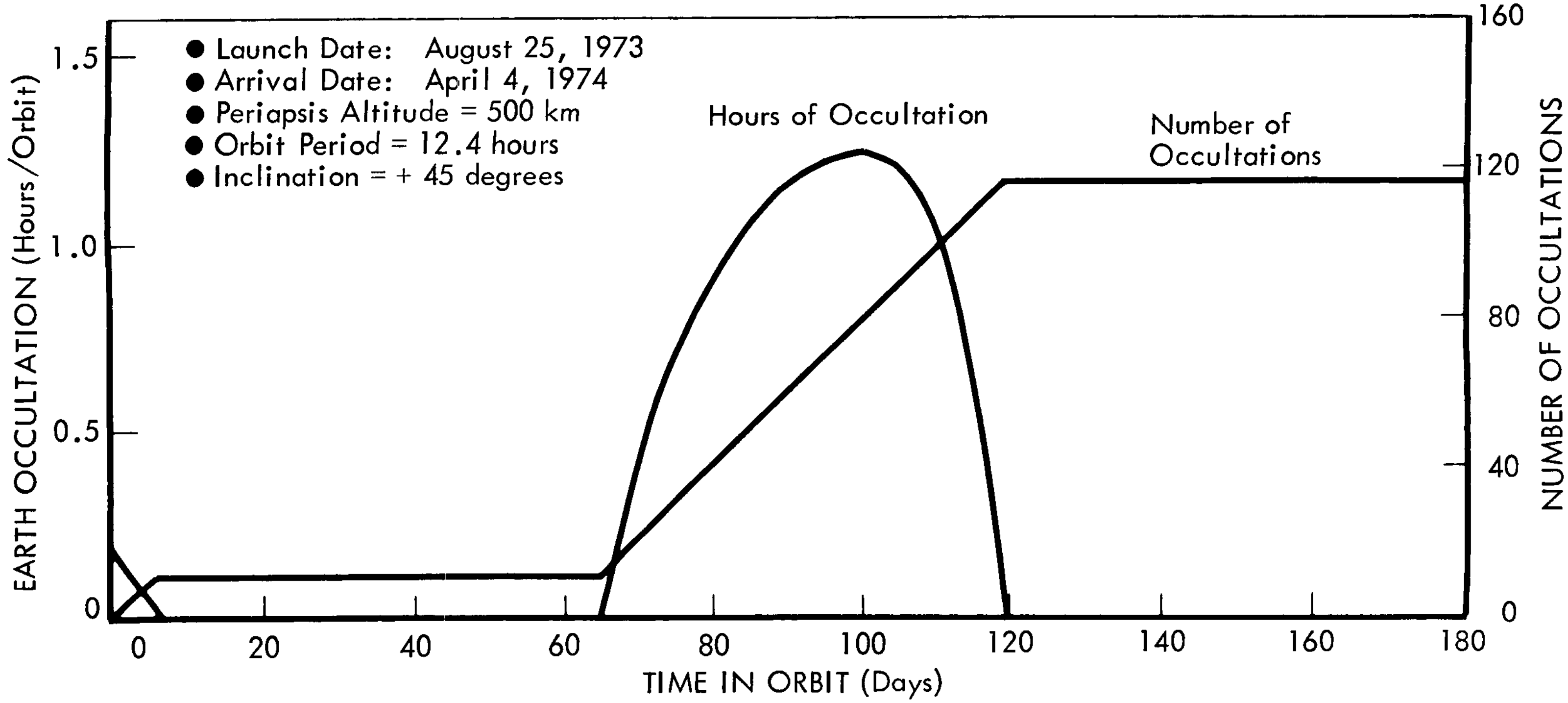


Figure 1.1.1-17: DESIGN ORBIT EARTH OCCULTATION

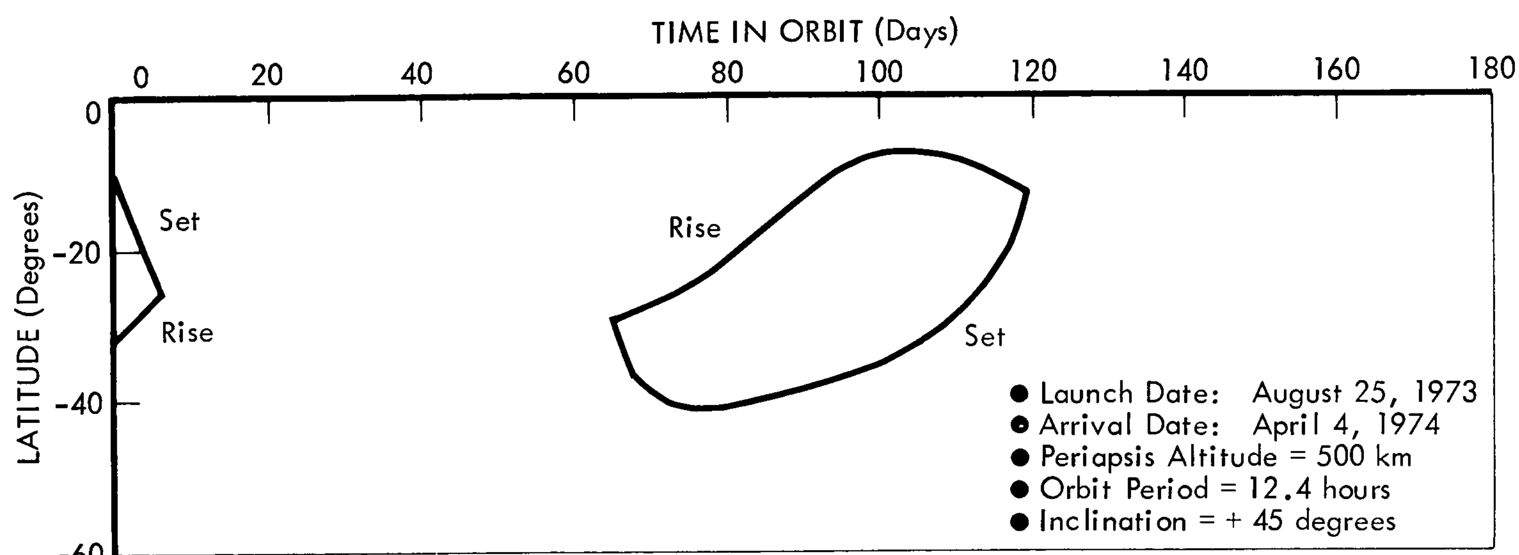


Figure 1.1.1-18: RISE AND SET LATITUDES — DESIGN ORBIT EARTH OCCULTATIONS

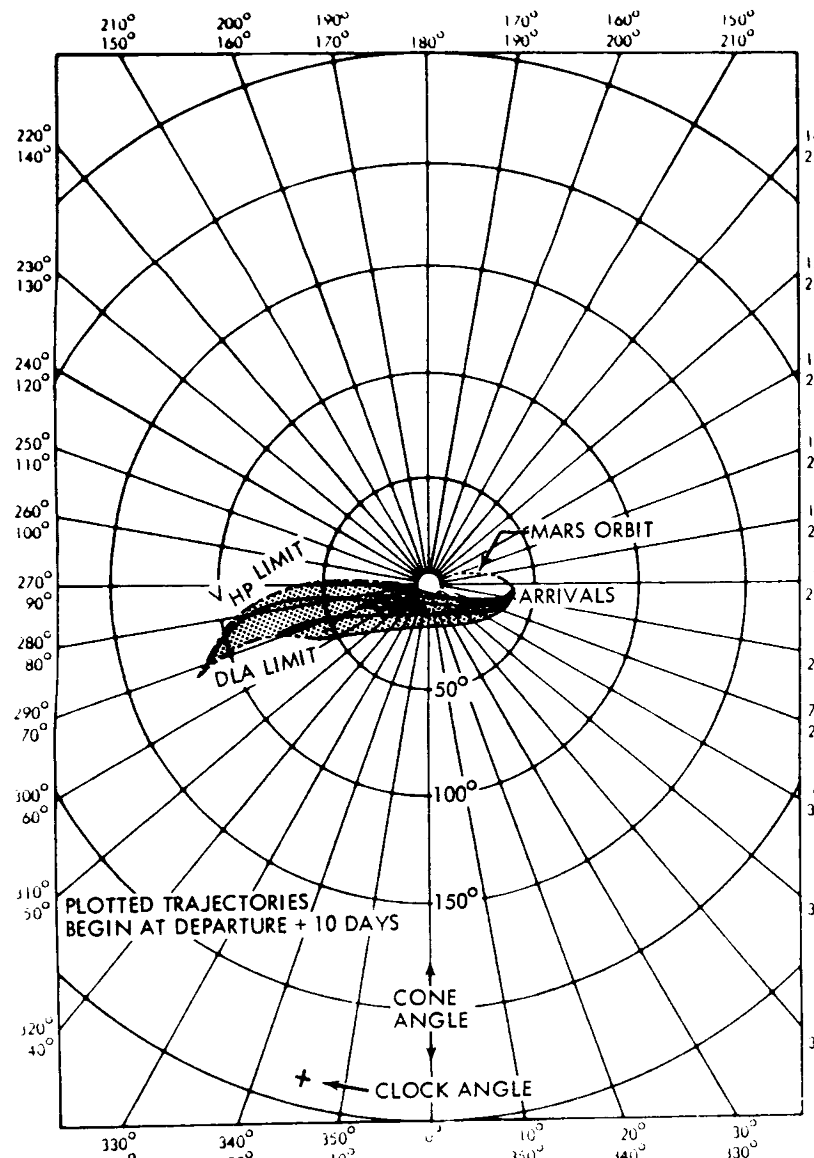


Figure 1.1.1-19: CONE & CLOCK ANGLES OF EARTH 1973 MARS TYPE I

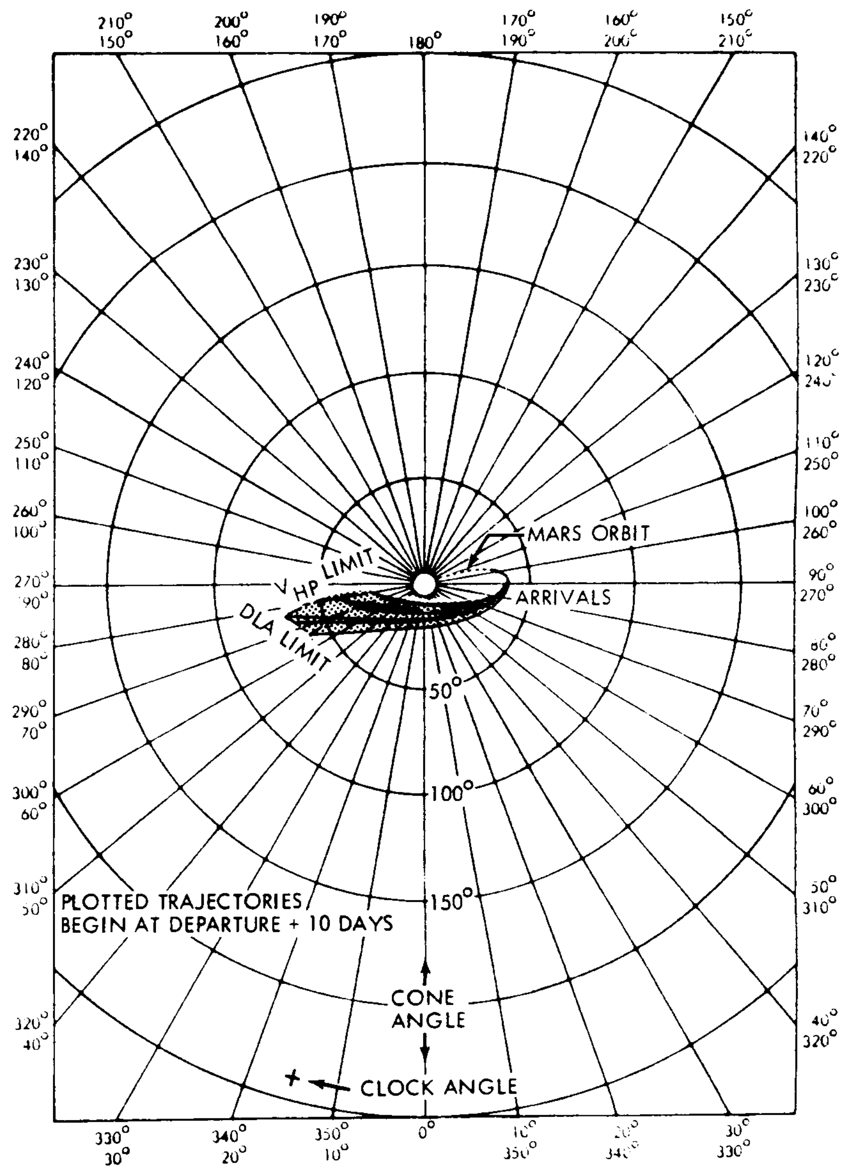


Figure 1.1.1-20: CONE AND CLOCK ANGLES OF EARTH 1975 MARS TYPE I

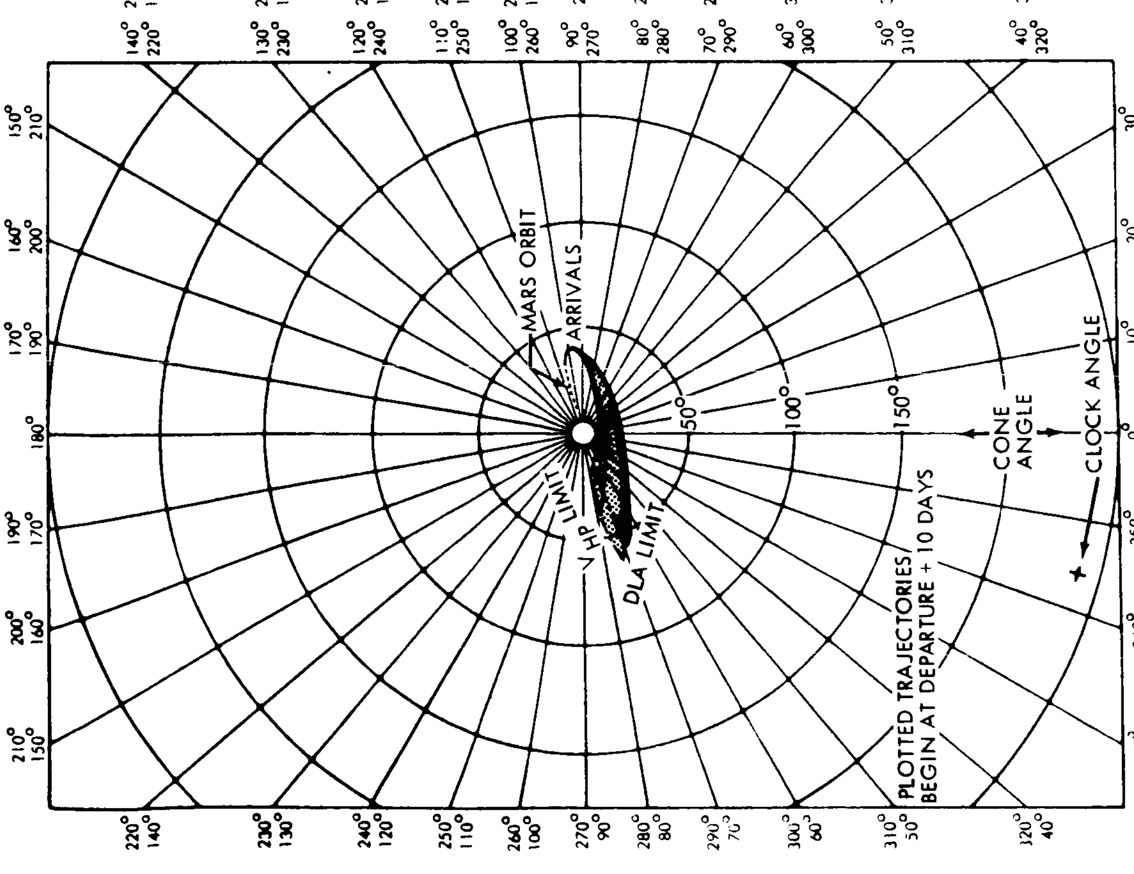


Figure 1.1.1-21: CONE AND CLOCK ANGLES OF EARTH 1977 MARS TYPE I

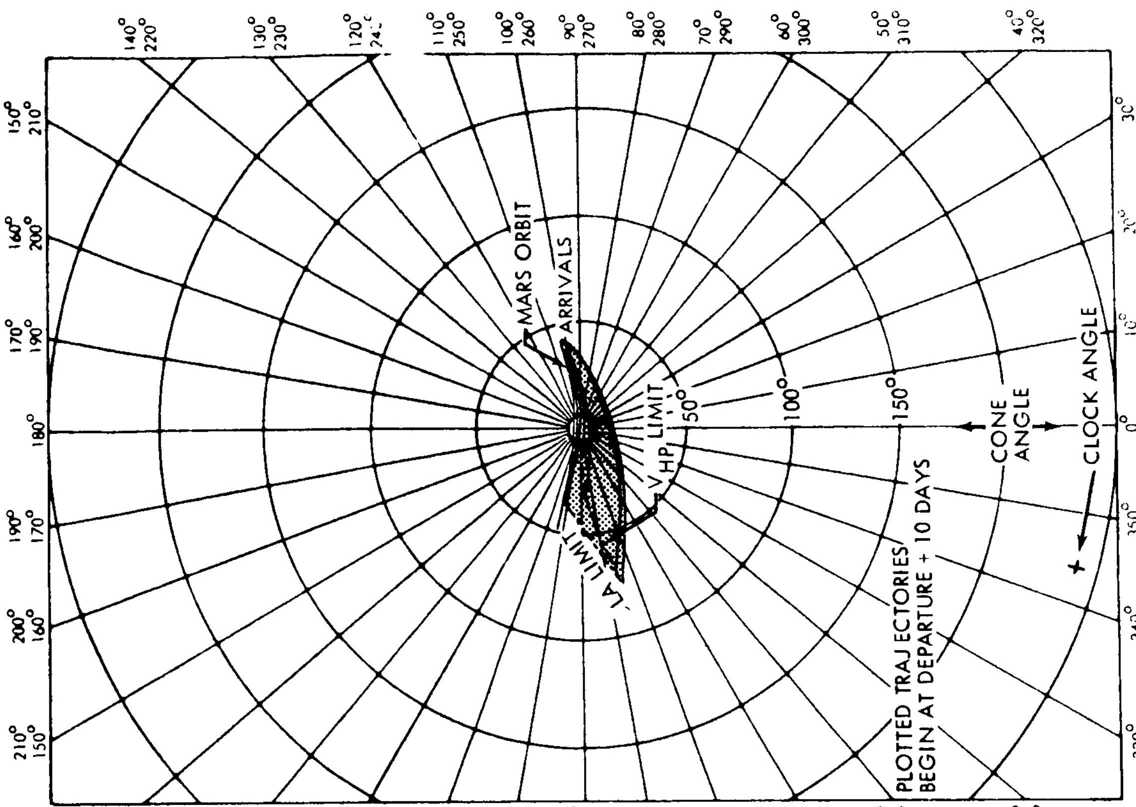


Figure 1.1.1-22: CONE AND CLOCK ANGLES OF EARTH 1979 MARS TYPE I

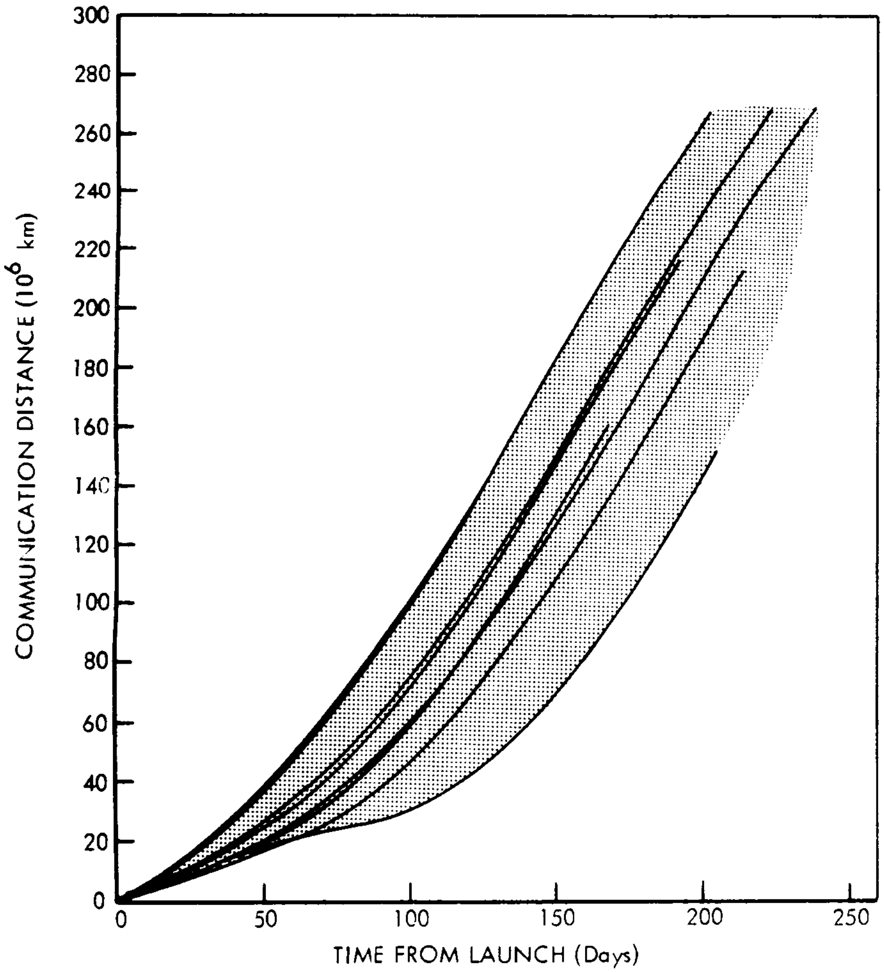


Figure 1.1.1-23: COMMUNICATION DISTANCE — 1973 MARS TYPE 1 TRAJECTORY ENVELOPE

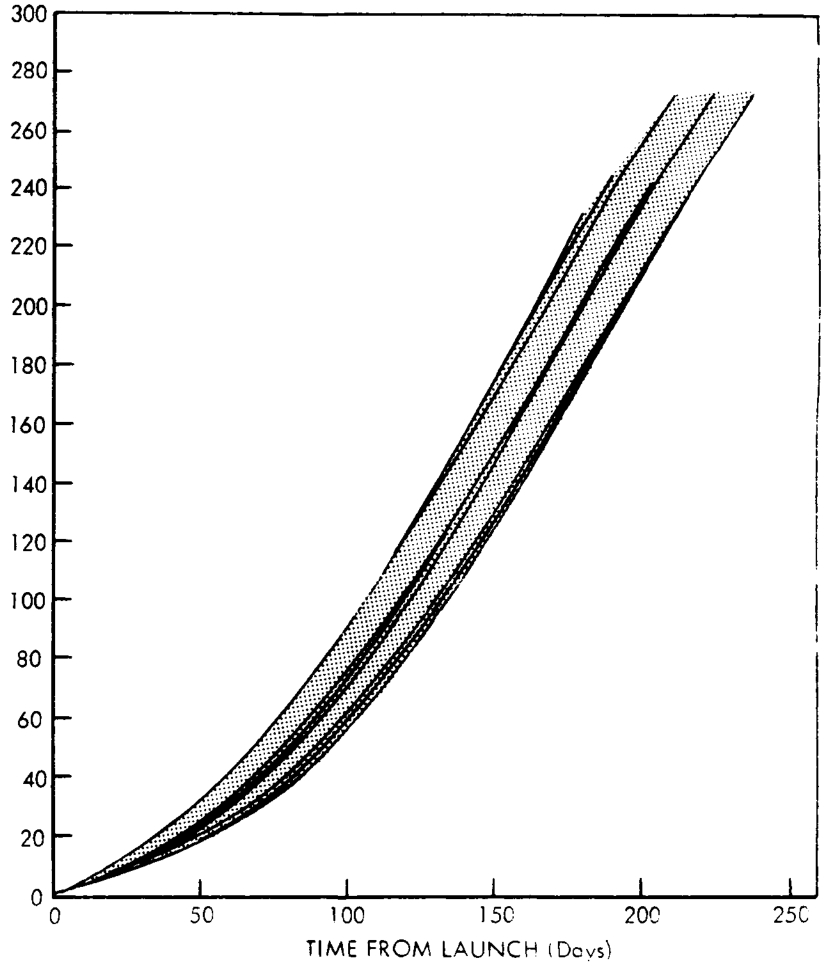


Figure 1.1.1-24: COMMUNICATION DISTANCE —1975 MARS TYPE I TRAJECTORY ENVELOPE

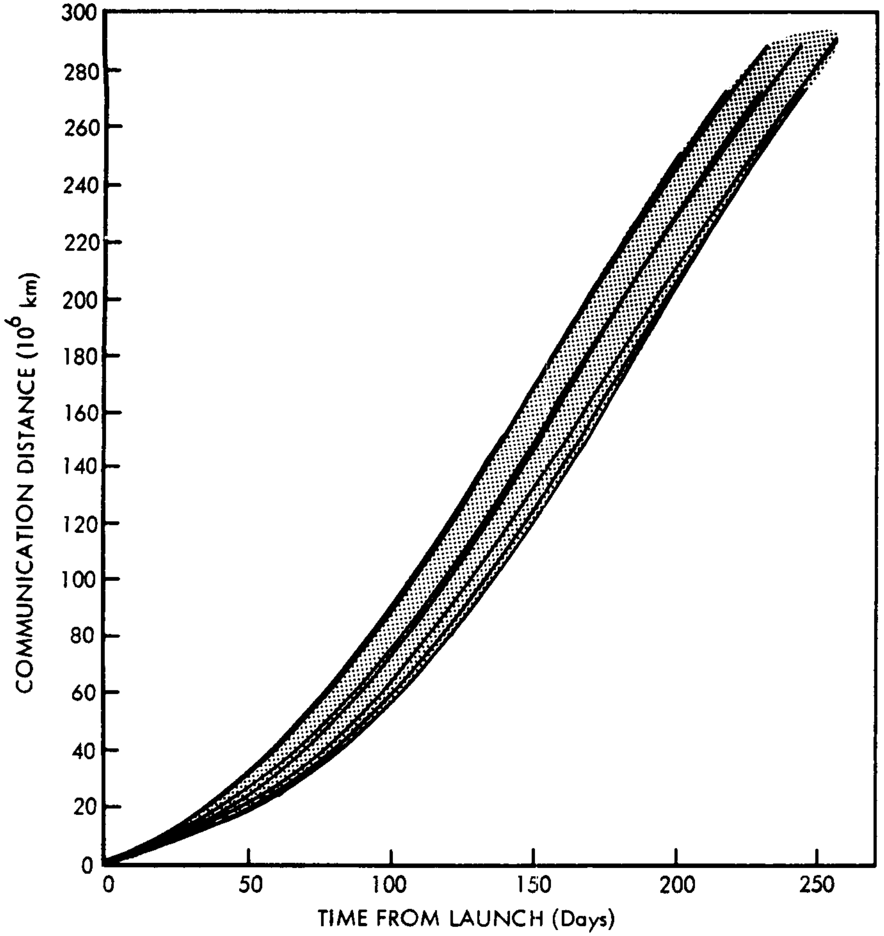


Figure 1.1.1-25: COMMUNICATION DISTANCE —1977 MARS TYPE I TRAJECTORY ENVELOPE

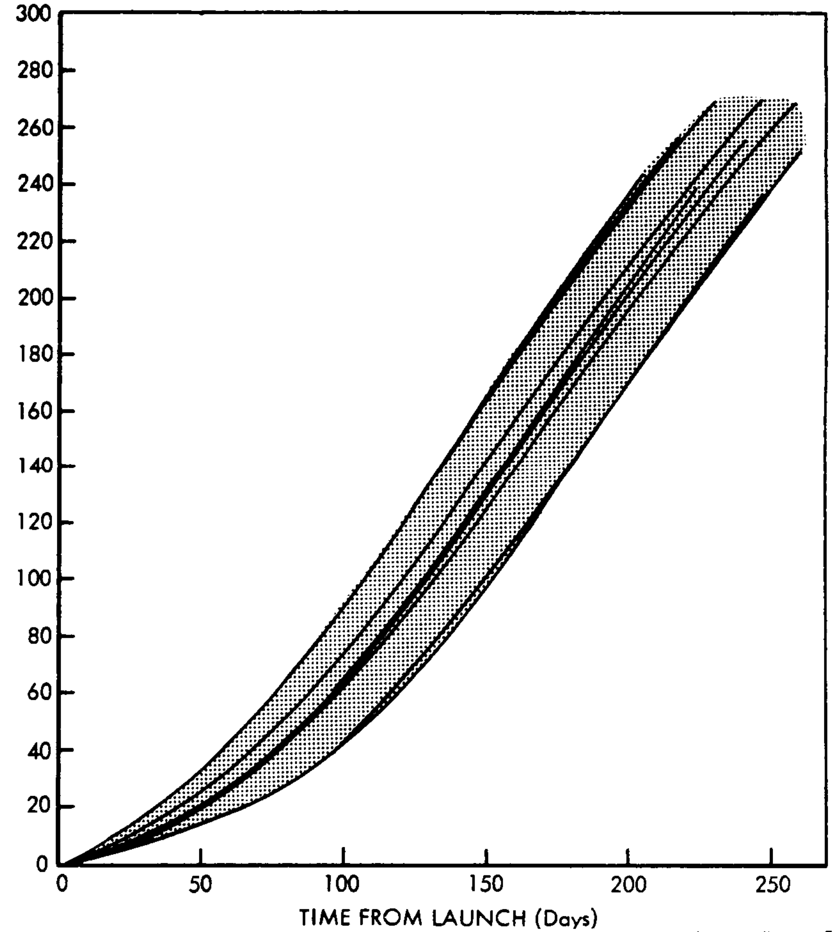


Figure 1.1.1-26: COMMUNICATION DISTANCE — 1979 MARS TYPE I TRAJECTORY ENVELOPE

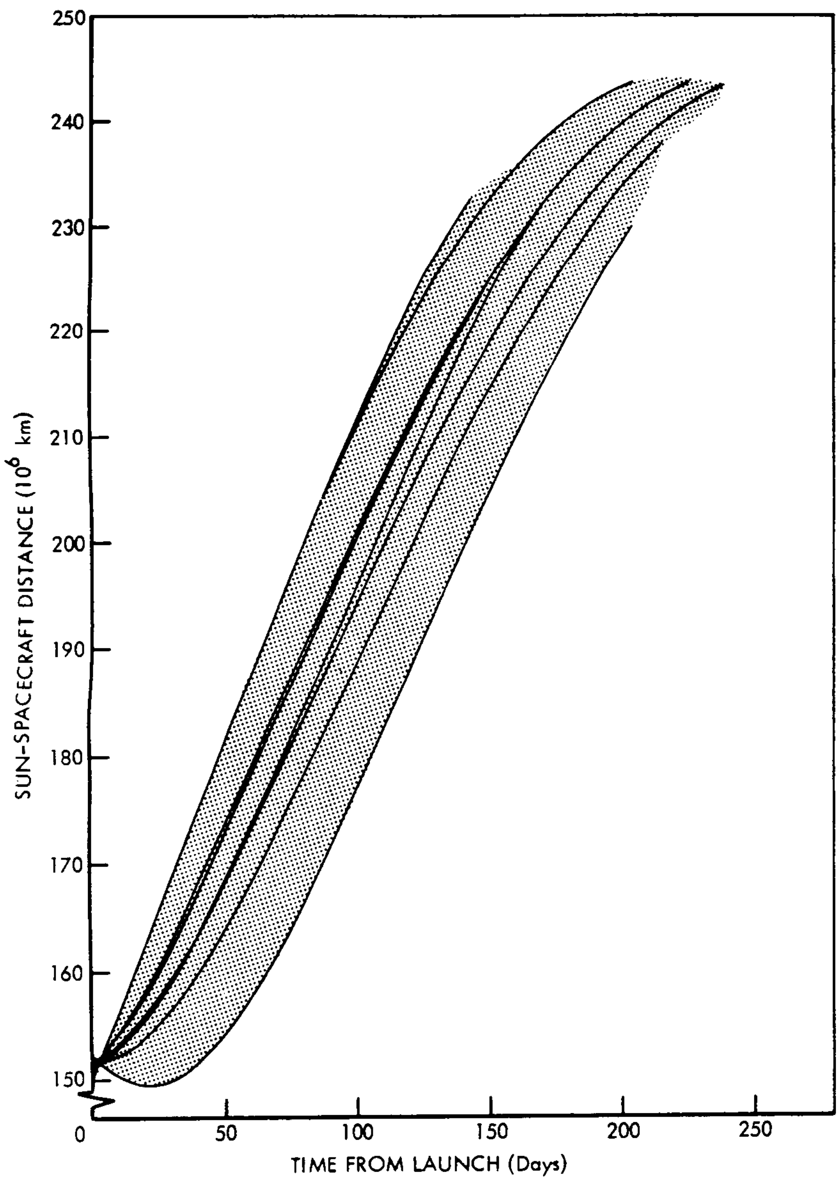


Figure 1.1.1-27: SUN-SPACECRAFT DISTANCE — 1973 MARS TYPE I TRAJECTORY ENVELOPE

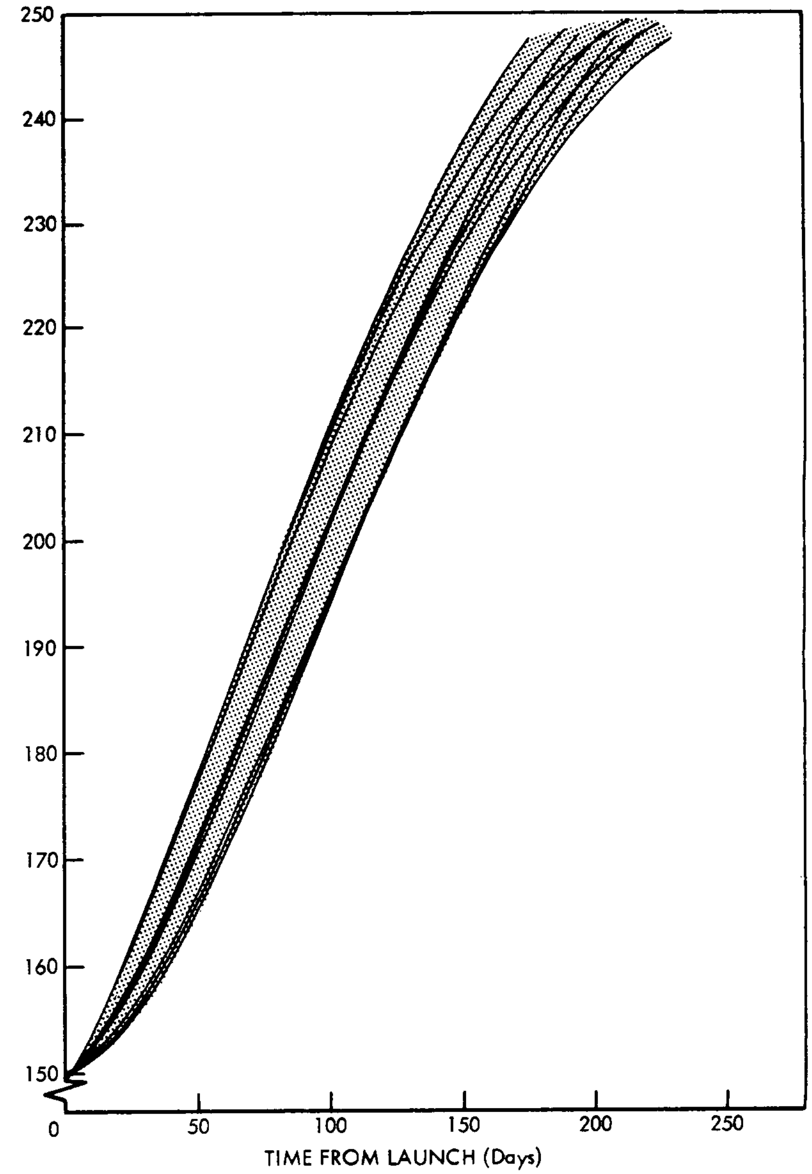


Figure 1.1.1-28: SUN-SPACECRAFT DISTANCE — 1975 MARS TYPE I TRAJECTORY ENVELOPE

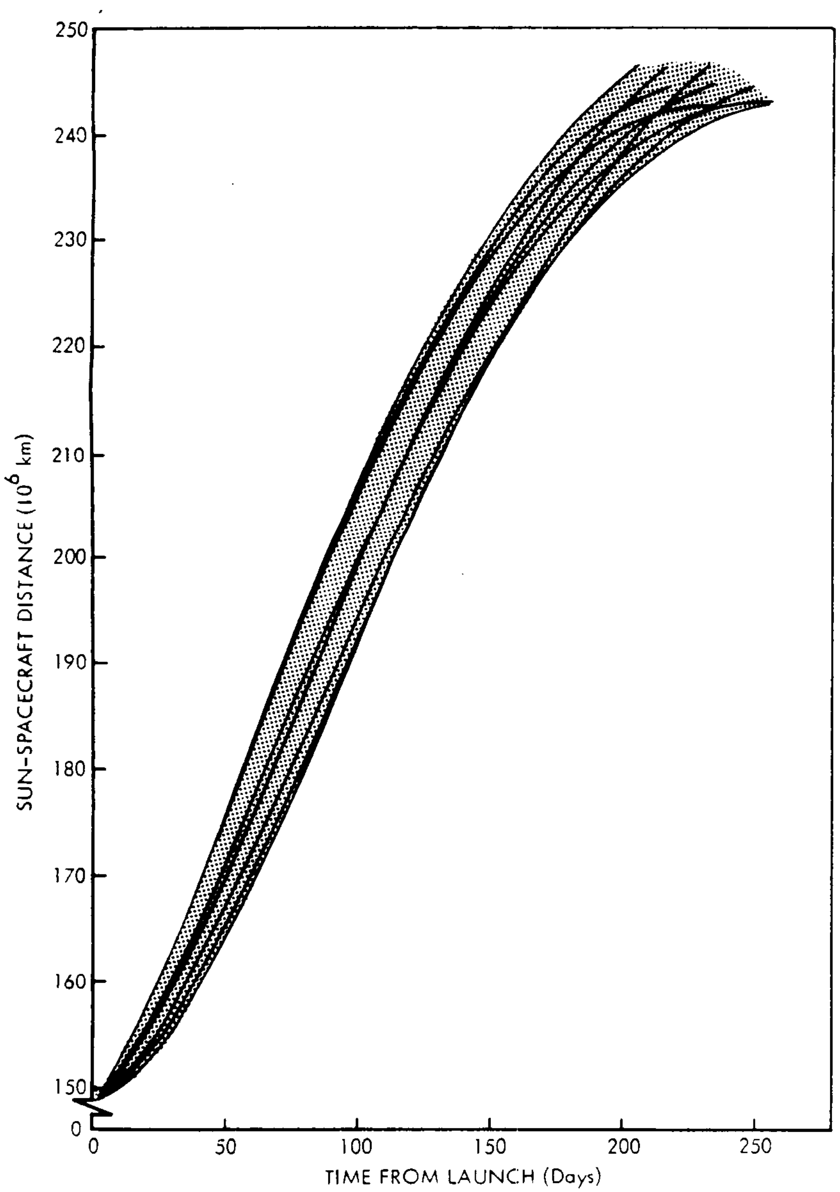


Figure 1.1.1-29: SUN-SPACECRAFT DISTANCE — 1977 MARS TYPE I TRAJECTORY ENVELOPE

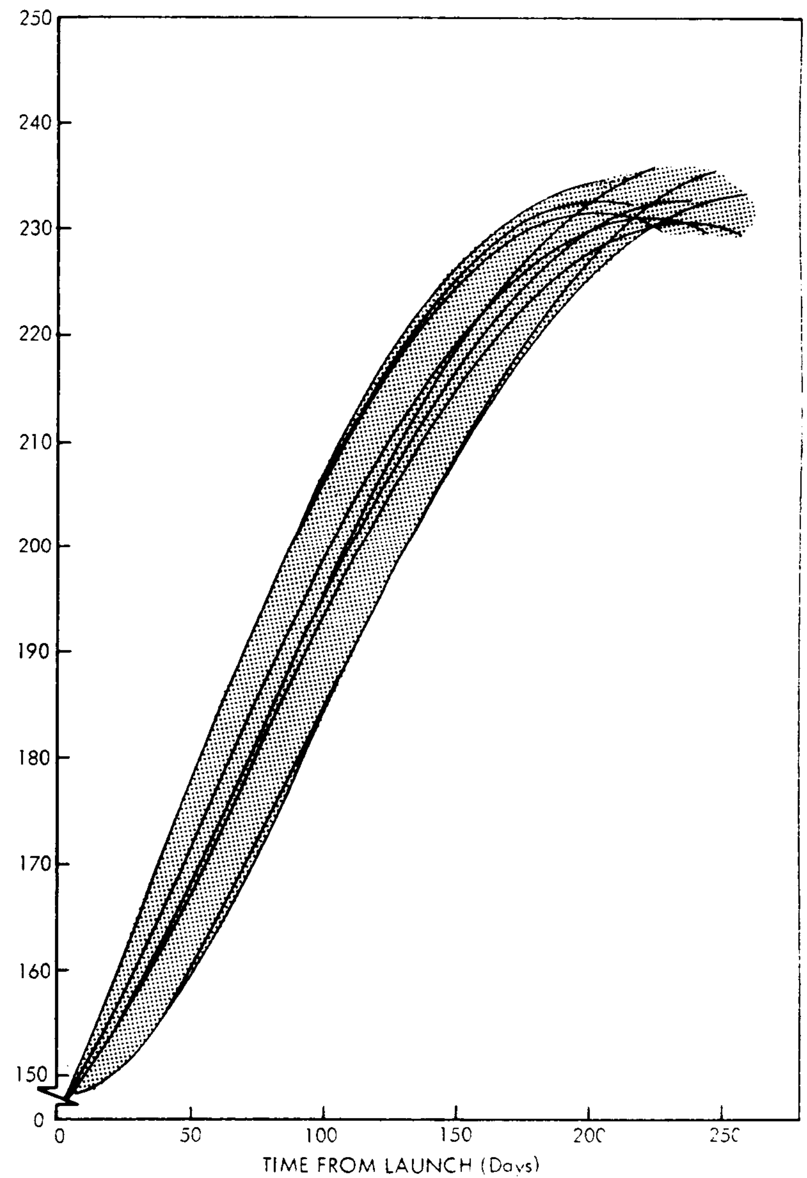


Figure 1.1.1-30: SUN-SPACECRAFT DISTANCE — 1979 MARS TYPE I TRAJECTORY ENVELOPE

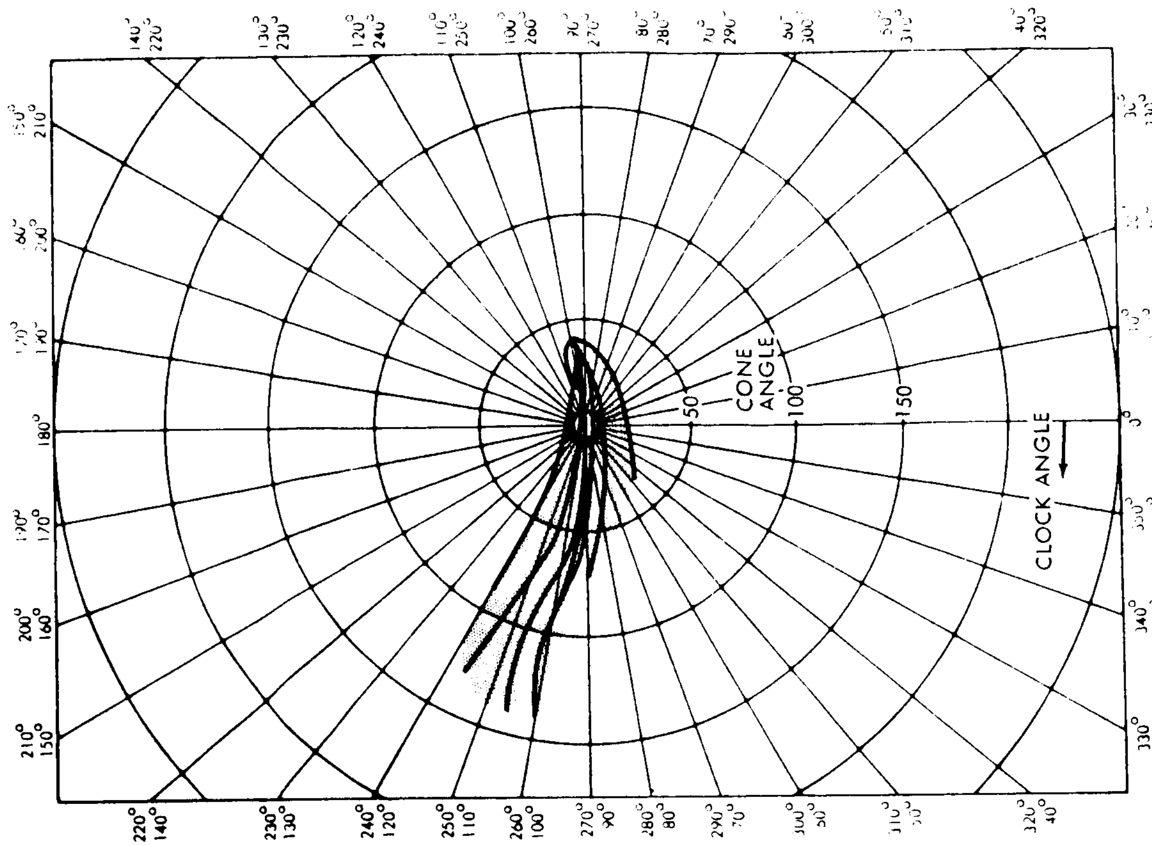


Figure 1.1.1-32: EARTH CONE ANGLE ENVELOPE 1977 TYPE II

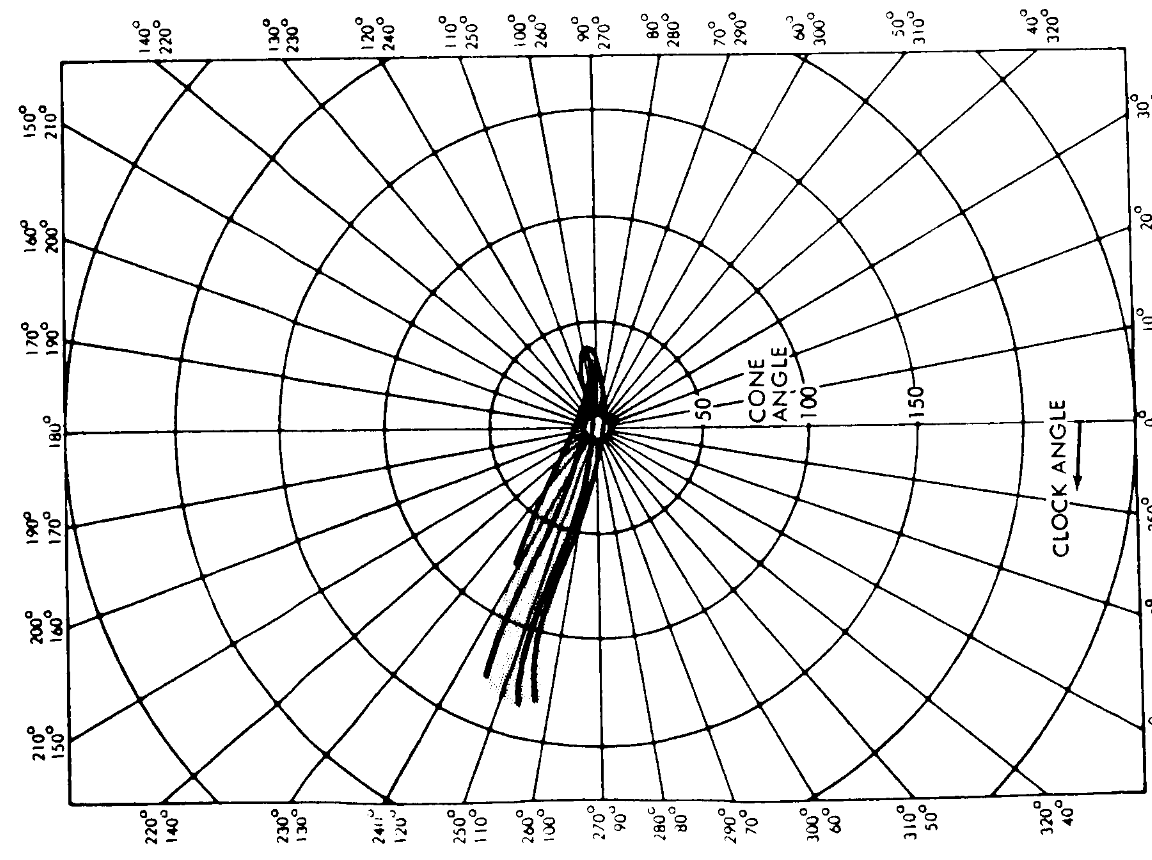


Figure 1.1.1-31: EARTH CONE ANGLE ENVELOPE 1975 TYPE II

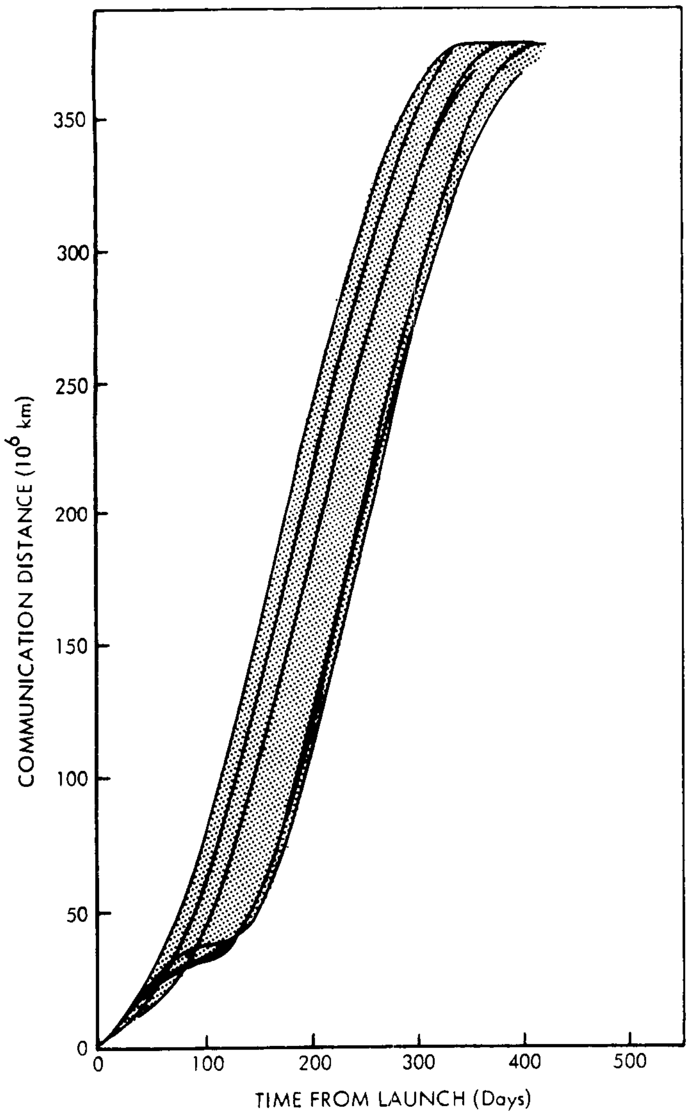


Figure 1.1.1-33: COMMUNICATION DISTANCE — 1975 MARS TYPE II TRAJECTORY ENVELOPE

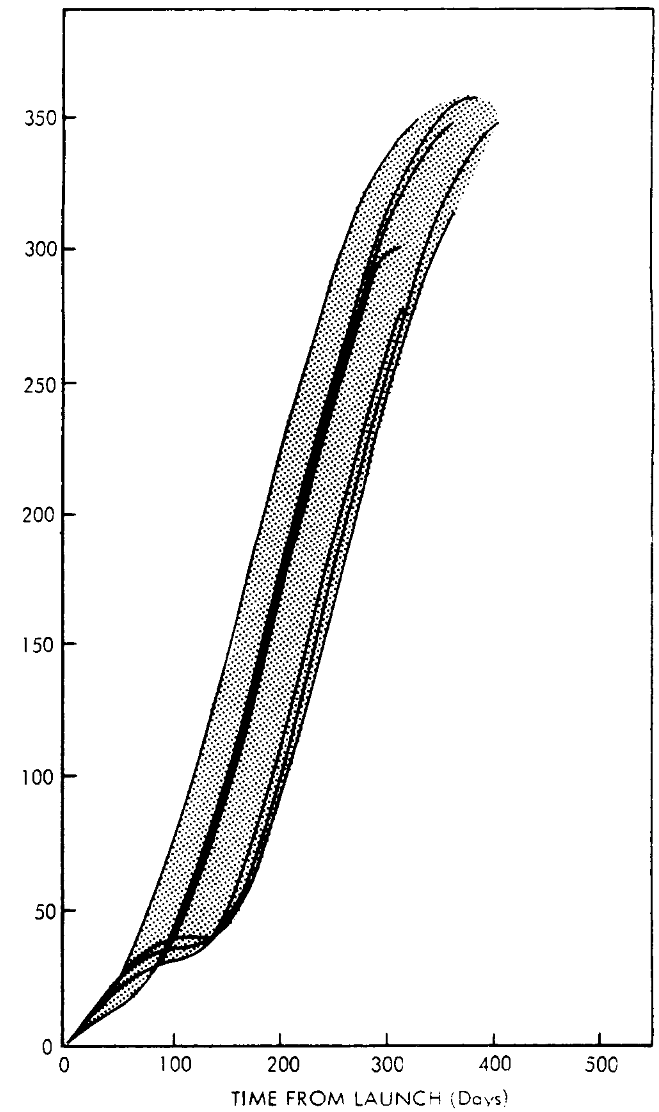


Figure 1.1.1-34: COMMUNICATION DISTANCE — 1977 MARS TYPE II TRAJECTORY ENVELOPE

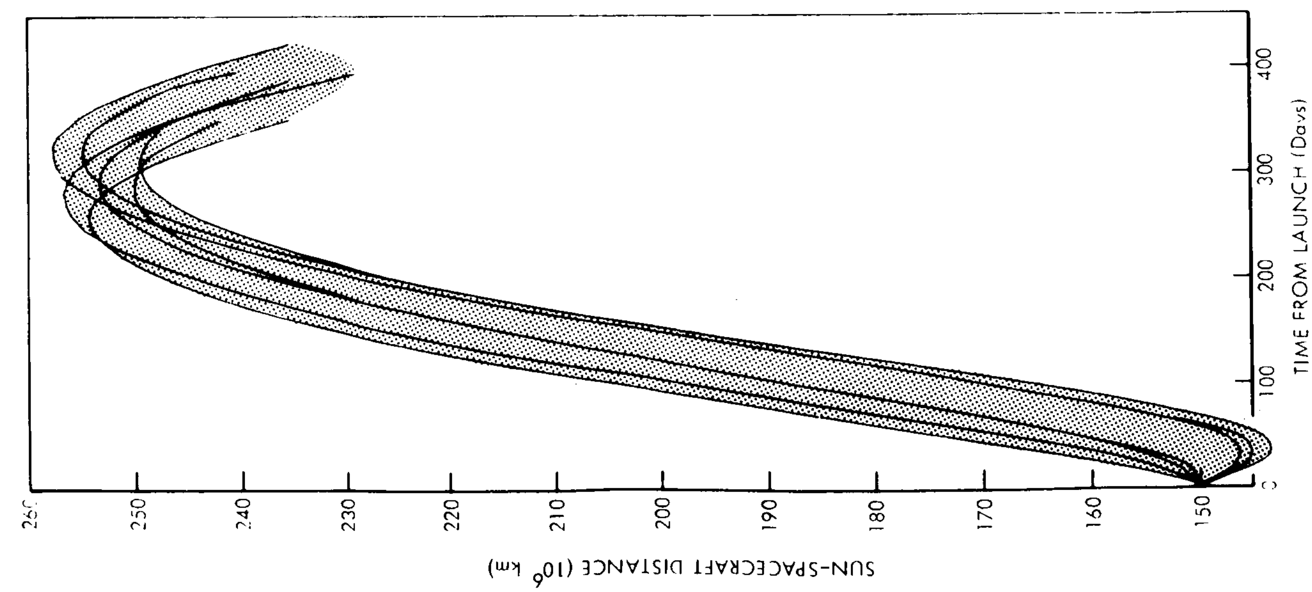


Figure 1.1.1-35: SUN-SPACECRAFT DISTANCE — 1975 MARS TYPE II TRAJECTORY ENVELOPE

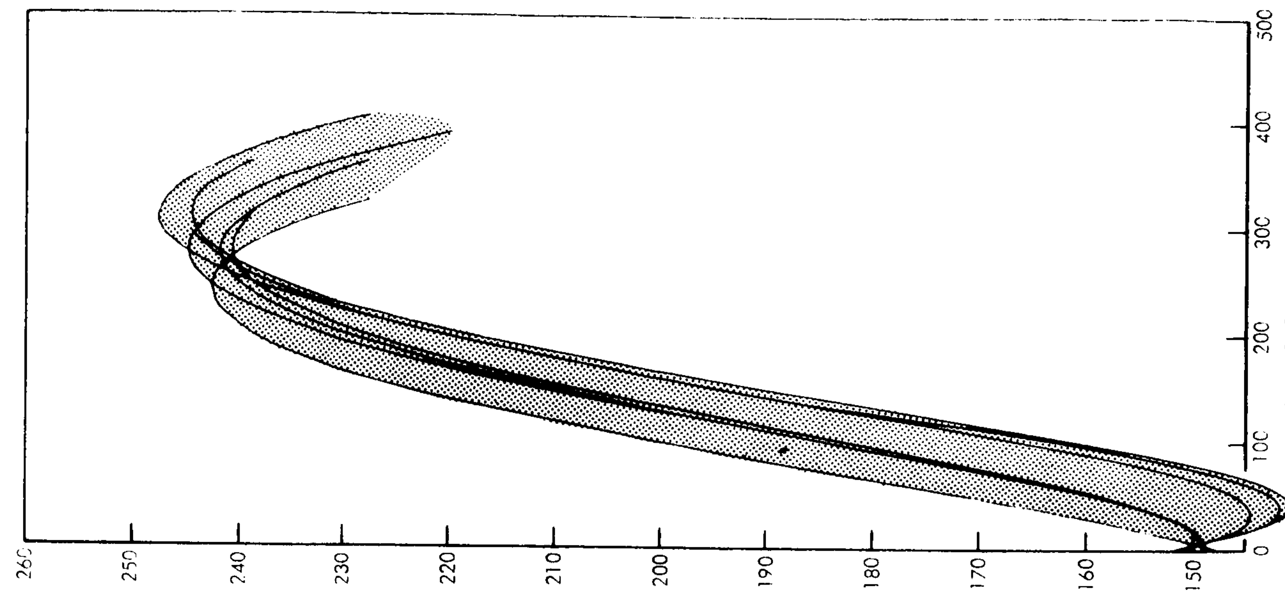


Figure 1.1.1-36: SUN-SPACECRAFT DISTANCE — 1977 MARS TYPE II TRAJECTORY ENVELOPE

1.1.2 Spacecraft Design Parameters

A summary of weight, reliability, and power requirements for the spacecraft and subsystem is shown in Table 1.1.2-1.

1.1.3 Spacecraft Functional and Subsystem Interfaces

This section describes the spacecraft interfaces with the launch vehicle system, capsule system, science subsystem, mission operations system, and tracking and data system. Figure 1.1.3-1 shows the major functional and subsystem interfaces. These interfaces are concerned primarily with the physical, electrical, and other hardware aspects of the interfacing systems. Software interfaces involving interface drawings, criteria documents, computer programs, test specifications, and standards are not discussed in this document.

1.1.3.1 Spacecraft-Launch Vehicle System Interface

Interfaces between the spacecraft and the launch vehicle system will provide the spacecraft systems with mechanical support, environmental control through air cooling and heat transfer, electrical power, and command and telecommunication data transmission. In addition, an interface will exist during oxidizer and fuel loading on the launch pad.

The mechanical interfaces are indicated in Figure 1.1.3-2. The mechanical interface is primarily for structural support. The shroud-spacecraft interface is constrained by shroud clearance requirements during separation. Dynamic envelopes 240 inches in diameter and 290 inches long are provided within the launch vehicle shroud for each of the two planetary vehicles. Four field joints are provided for assembly of the shroud. An 18- x 21-inch annular space next to the shroud below each planetary vehicle is reserved for instrument unit cabling. Each planetary vehicle is attached to the shroud at eight places.

The shroud separation joint locations recommended in the Task D guidelines document are shown at some distance above the planetary vehicle attachment points. A study will be necessary to optimize the relationship of shroud separation joint location with respect to planetary vehicle attachment point. This study should consider planetary vehicle shroud clearance, tipoff rates, and separation velocities.

Cooling air for environmental control at the launch pad will be provided from the environmental control facilities. To enable transition to the thermal control required during launch, the electronic assembly radiator plates must be cooled to a temperature 50°F or less at launch. Approximately 2100 scfm cooling air at 40°F with dewpoint below 35°F and with Class 100,000 clean room conditions will be ducted to each spacecraft through an umbilical to the shroud.

A flow-operated switch will be required to provide warning in the event of air flow failure.

Preliminary investigations have indicated the possibility of damage to the louver assemblies due to flutter induced by the cooling air flow. In this event, an integral liquid coolant loop built into the radiator plates may be preferable.

Table 1.1.2-1: SPACECRAFT SUBSYSTEM CHARACTERISTICS SUMMARY

SUBSYSTEM	SUBSYSTEM WEIGHT (1)	RELIABILITY		POWER
		ALLOCATION	ASSESSMENT	
Spacecraft Bus				
Power	539	0.991	0.989	See Table 1.2.1-2
Guidance and Control	213	0.983	0.957	64.6
Data Storage	134	0.923	0.992	54.0
Telemetry	34	0.995	0.995	16.9
Radio	100	0.962	0.964	223.0
Antenna	125	0.995	0.999	3.0
Computing and Sequencing	69	0.942	0.959	68.8
Propulsion	2077	0.995	0.996	9.0 (2)
Structural and Mechanical	1031	0.936	0.936	0
Pyrotechnic	59	0.9999	0.9999	11.8
Temperature Control	184	0.9999	0.9999	0
Cabling	265	0.994	0.994	0
Contingency (Weight)	257			-
Science Subsystem	482	-	-	45.0 (163 for 1 Hr)
Flight Spacecraft Total (inert)	5569	(3) 0.747	(3) 0.799	904

(1) Subsystem weight as defined for reliability analysis only. Detail weight per NASA format is shown in Table 1.1.7-1

(2) Excludes short term maneuver loads.

(3) Excludes Science Subsystem Reliability

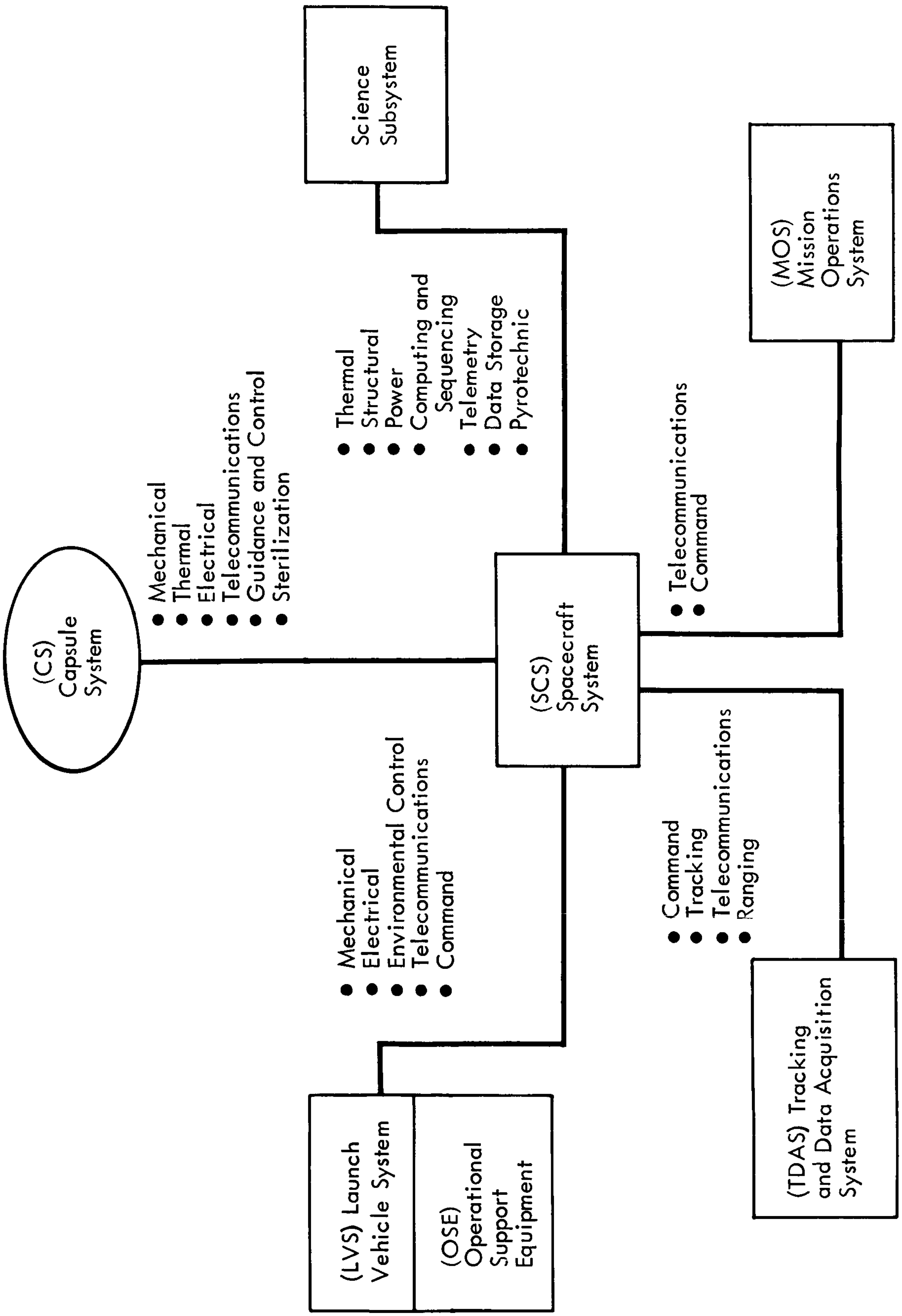


Figure 1.1.3-1: SPACECRAFT FUNCTIONAL AND SUBSYSTEM INTERFACES

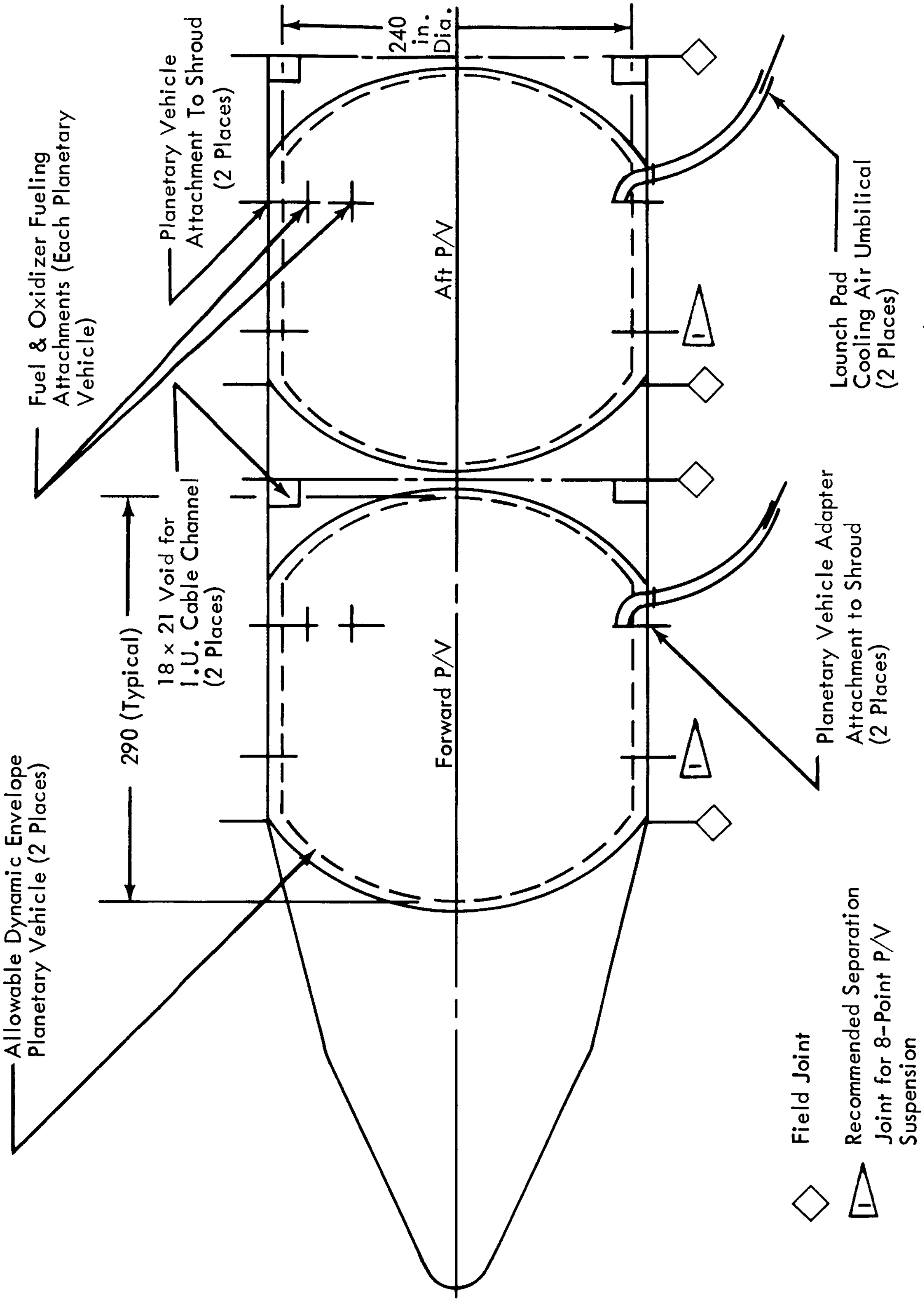


Figure 1.1.3-2: PHYSICAL INTERFACES — SPACECRAFT/LAUNCH VEHICLE SYSTEM

Spacecraft thermal control during launch and boost phase will be by heat transfer through the launch vehicle/shroud interface. During boost, heating of the spacecraft by the shroud must be limited to 65 watts/ft^2 maximum, measured at the spacecraft. This requirement implies that the interior of the shroud must be insulated and covered with a low emittance coating ($\epsilon < 0.10$). The insulation requirement will be greatest in the ogive section of the shroud where aerodynamic heating will be higher.

Electrical power interface requirements exist during launch pad operations and during the mission boost phase (up through Earth orbit to Mars transit insertion). During pad operation, power will be supplied through umbilicals from the launch vehicle support system in the mobile launcher. A maximum of 900 watts at 70 volts d.c. will be required. Figure 1.1.3-3 shows the electrical interfaces between the spacecraft and the launch vehicle system.

An umbilical cable to each planetary vehicle provides for power, RF command and telemetry signals, and prelaunch checkout on the launch pad. Planetary vehicle telecommunications and commands interface with the launch vehicle instrument unit. Before launch, the command and telemetry communications link with the ground equipment utilizes three redundant paths. The unmodulated pulse code modulation data train is distributed both through the planetary vehicle umbilical to the launch control equipment and to the launch vehicle instrument unit for modulation of the UHF transmitter. In addition, the planetary vehicle S-band launch transmitter is diplexed through separable cabling to the S-band transponder antennas in the launch vehicle instrumentation unit.

During launch, planetary vehicle communication with the ground tracking stations exists through the instrument unit UHF link and through the S-band planetary vehicle launch transmitter diplex connection to the instrument unit antennas.

A fueling service interface will exist during the countdown for loading fuel and oxidizer to the spacecraft in a payload stack on the launch pad. Fueling will be accomplished by manual connections from the mobile service structure hypergolic fuel transfer system. The spacecraft propellant tanks will be loaded with approximately 4300 pounds of monomethyl hydrazine and 6900 pounds of nitrogen tetroxide for the 1973 opportunity. Spacecraft environment will be retained at Class 100,000 or better clean room conditions, and fuel and fuel line particulate contamination will be held within specified particulate absolute values.

Two boxes will be located in each shroud for each spacecraft, one for fuel and one for oxidizer filling. The boxes will form a physical and biological barrier between the outside and the inside of the shroud containing the spacecraft and planetary vehicle (see Figure 1.1.3-4). The box will be located below the planetary vehicle so it will not interfere with the physical separation. It will contain fittings for the fill line, the drain line, and the overflow line from the fuel tanks. On the spacecraft side of the box there will be a separation joint to separate the fill and overflow lines from the spacecraft when the spacecraft leaves the shroud.

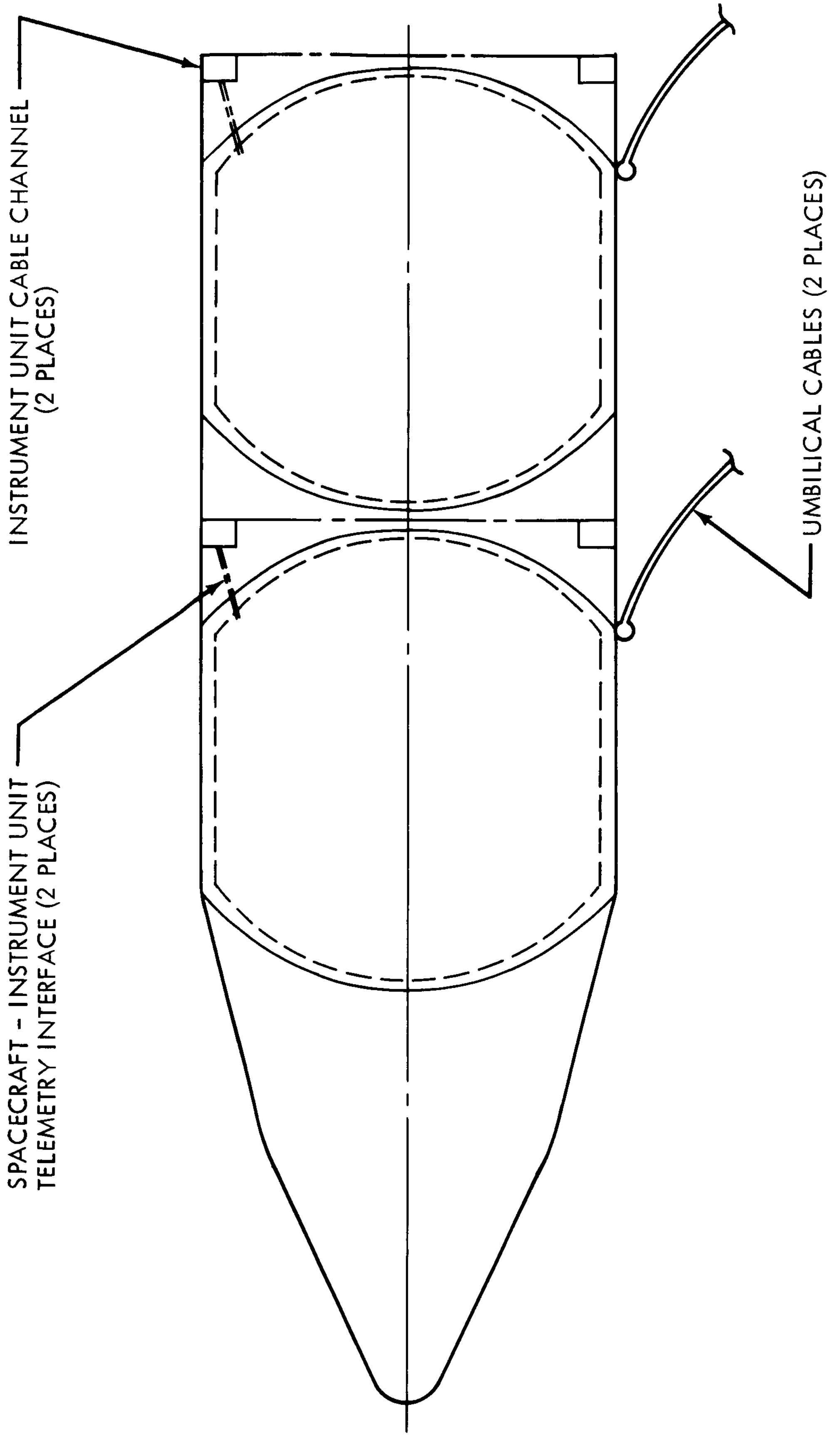


Figure 1.1.3-3: ELECTRICAL INTERFACES — SPACECRAFT / LAUNCH VEHICLE SYSTEM

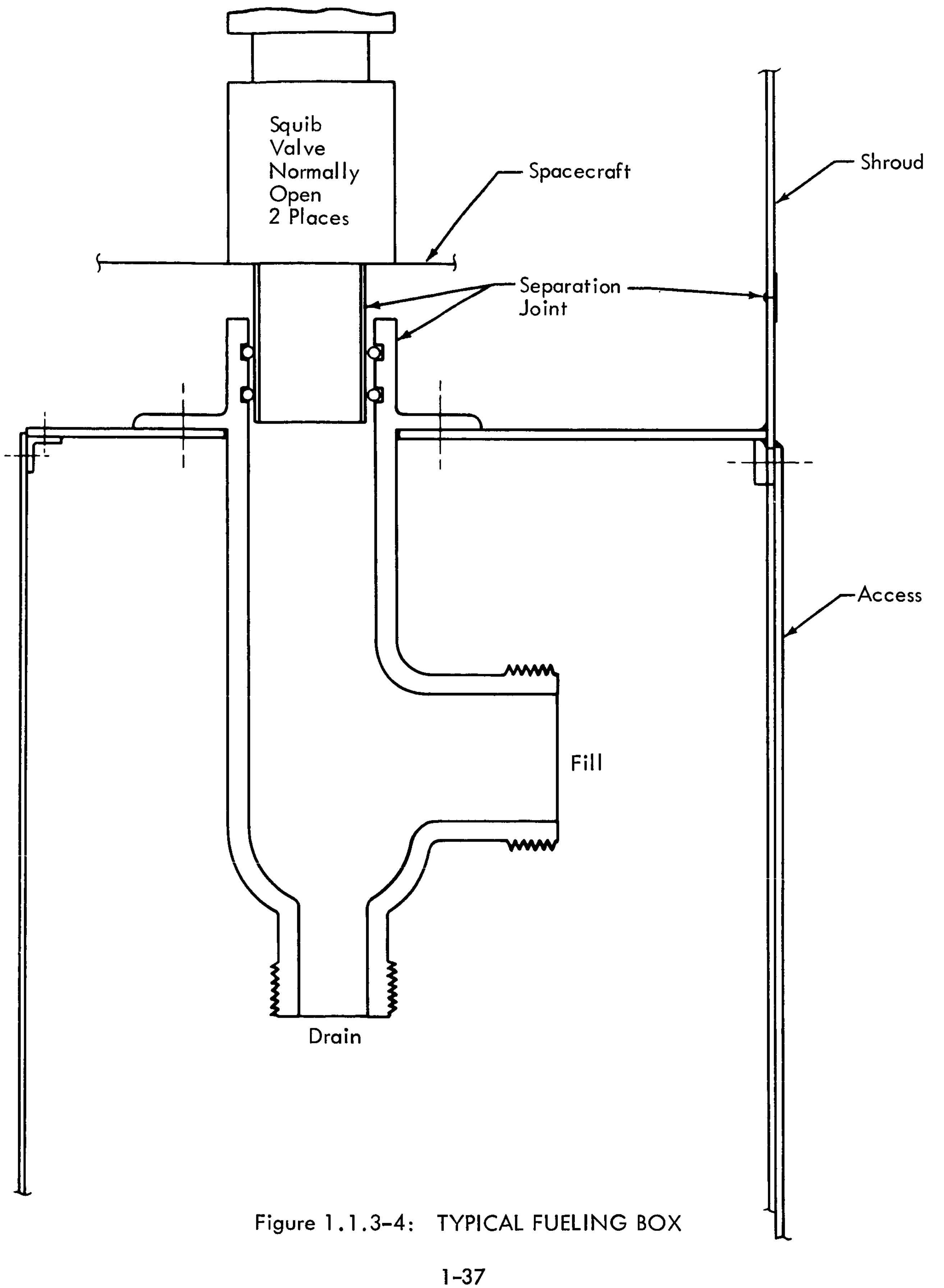


Figure 1.1.3-4: TYPICAL FUELING BOX

The reference axes as applied to the Voyager spacecraft are shown in Figure 1.1.3-5. Figure 1.1.3-6 shows the relationship of the axes with respect to the launch vehicle. Figure 1.1.3-7 shows the axes relationships as related to the ground and launch pad.

1.1.3.2 Spacecraft-Capsule System Interfaces

Known interfaces with the capsule have been identified, and capsule-spacecraft interface problem areas discussed.

Mechanical -- The structural-mechanical interface with the flight capsule is a 160-inch diameter multibolt joint. The structural interface is designed to carry a 7000-pound flight capsule under the combined load conditions **shown in Section 1.2.10.**

The total planetary vehicle center of mass offset uncertainty is a major factor in determining the thrust vector control pointing accuracy. Consequently, the spacecraft and capsule center of masses must be controlled, together with the tolerances of the spacecraft capsule field joint, to provide an acceptable combined center-of-mass offset uncertainty. The acceptable volume of uncertainty for the planetary vehicle center of mass may be defined as the frustrum of a cone lying along the vehicle X axis with its apex at the main propulsion engine gimbal point. The distance between the gimbal point and the top of the frustrum and the radius of the frustrum at that point are defined by the thrust vector control pointing error requirements.

Thermal -- The thermal barrier between the spacecraft and the flight capsule when the capsule is attached and between the spacecraft and space when the capsule is separated is the capsule lower sterilization cannister. The thermal control of the propulsion subsystem is dependent on a low insulation conductance at the spacecraft-capsule interface. The requirements on the insulation conductance of the lower sterilization canister is less than 0.90 watts/m^2 - ^{o}K. This value includes the effects of penetrations, electrical, and instrumentation feed-throughs, etc. That portion of the lower sterilization canister facing the propulsion subsystem also requires a highly reflective coating (aluminized mylar or equivalent) to radiatively couple the propulsion components to one another.

The spacecraft-capsule field joint is a significant heat leak to space. Once the spacecraft and capsule are mated, this joint will never be broken and can be well insulated. The insulation conductance requirement of the spacecraft-capsule field joint is 0.90 watt/m2 -^{o}K.

That portion of the lower sterilization canister outside the spacecraft-capsule attachment ring must have a thermal control coating that is compatible with the spacecraft. The coating requires a high diffuse emittance in the infrared spectrum and a high absorptance in the solar spectrum. The specific requirements are an α_s/ϵ of 1.0 $\pm$ 0.1 with an emittance of 0.9 $\pm$ 0.05.

Electrical and Telecommunications -- The capsule interfaces electrically with the spacecraft power, computer and sequencer, and the telemetry subsystems. The number of wires that cross these interface are shown below:

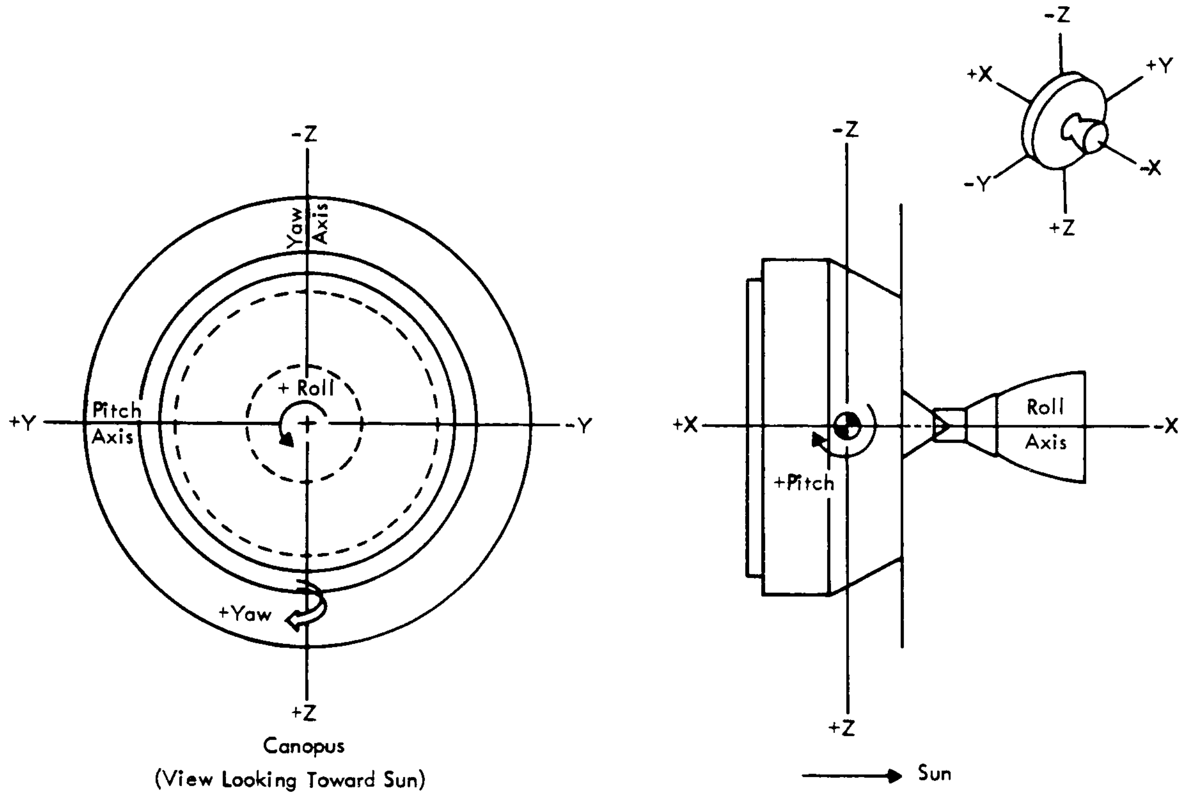
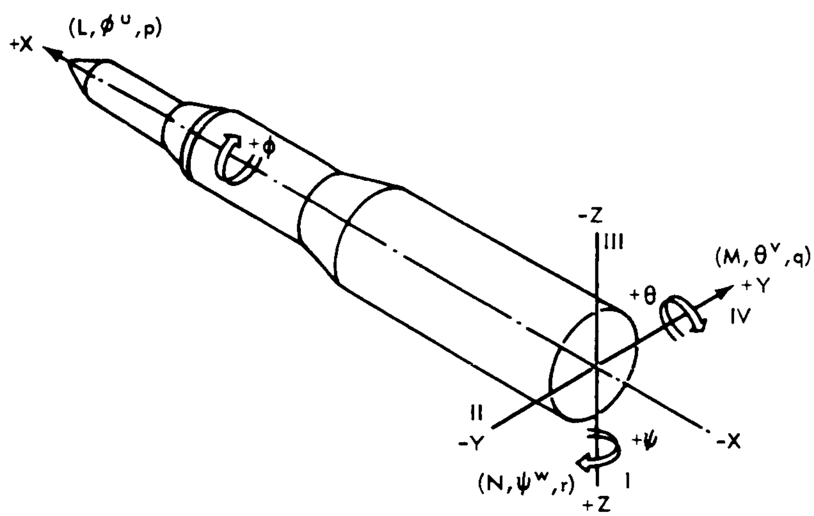


Figure 1.1.3-5: SPACECRAFT REFERENCE SYSTEM



AXIS	SYMBOL	SYMBOL	POSITIVE DIRECTION	FORCE SYMBOL	ANGLE	SYMBOL	LINEAR VELOCITY	ANGULAR VELOCITY
Longitudinal	X	L	Y to Z	X	Roll	ϕ	u	p
Lateral	Y	M	Z to X	Y	Pitch	θ	v	q
Vertical	Z	N	X to Y	Z	Yaw	ψ	w	r

Figure 1.1.3-6: LAUNCH VEHICLE COORDINATE SYSTEM

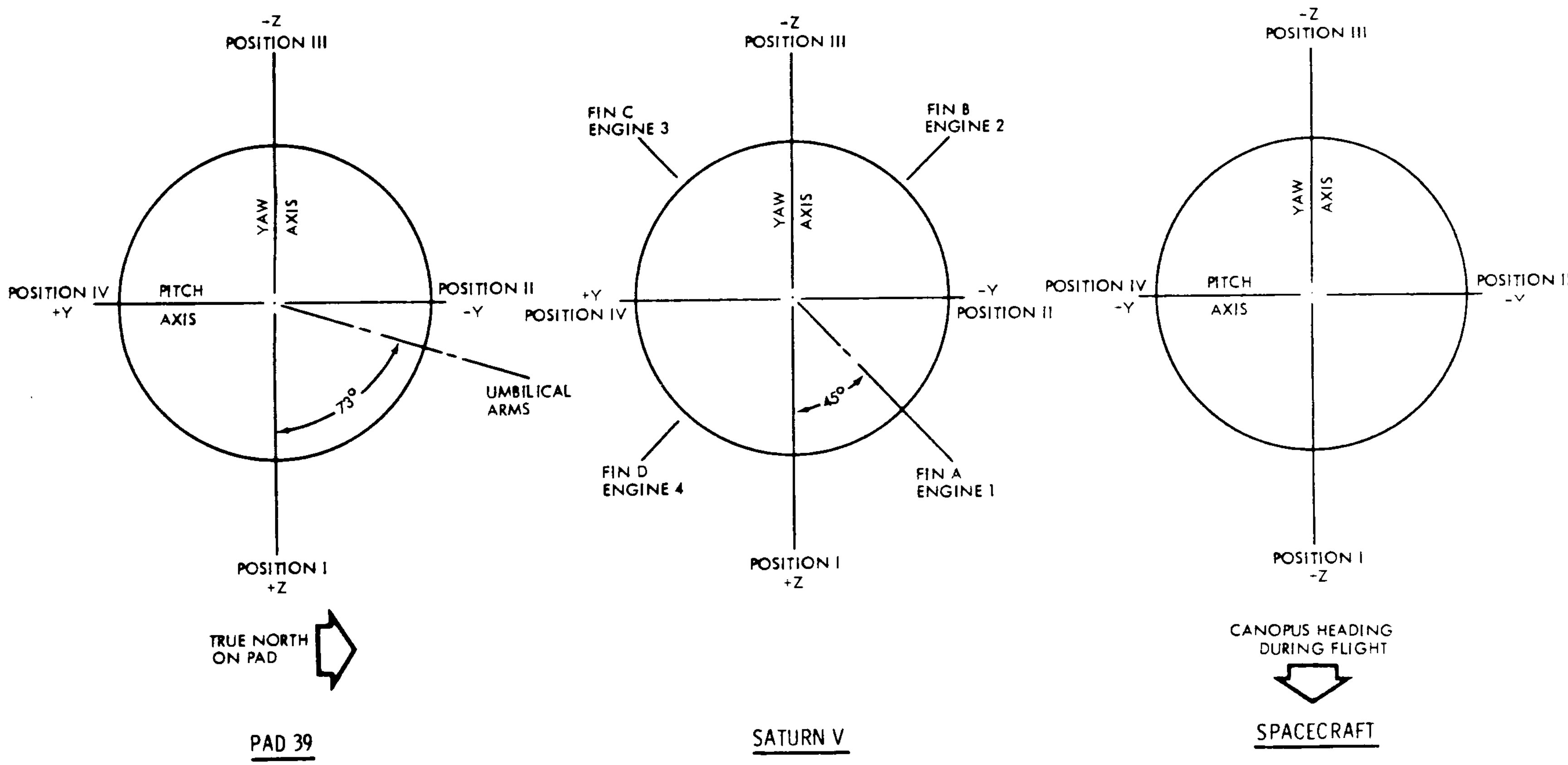


Figure 1.1.3-7: GENERAL POSITION DIAGRAM
(Looking Down Towards Ground at Launch or Towards Sun in Flight)

Subsystem	Wires
Power	4
C&S	33
Telemetry	8
Total	45

The function of each wire is shown in Figure 1.1.3-8. A 60-pin connector is used to handle the 45 wires allowing a 33% growth factor. When attached to the spacecraft, the capsule receives 200 watts of prime power, 12 different commands, and a 4.8-kHz clock signal. The capsule delivers telemetry data at either 5.8, 9.2, or 115.5 bps (depending on selected spacecraft telemetry mode) over the same interface.

After release, the capsule "interfaces" with two RF antennas and a relay receiver, described in Sections 1.2.6 and 1.2.7. A high data rate of 50 to 200 K bps is received during entry on the helix array and a 100 bps rate on the dipole antenna. The postlanding data rate of 1000 bps is received on the dipole.

Guidance and Control -- The following considerations impose a precise orientation requirement on the capsule-spacecraft interface:

1) Initial alignment conditions between the spacecraft celestial references and the capsule reference system may contribute to landing dispersions.

2) Initial conditions may affect requirements on capsule attitude sensors and control references in terms of dynamic range.

3) Initial orientation of the capsule affects its maneuver requirements.

4) Separation impulse effects on both capsule and orbiting spacecraft must be accounted for in predicting relative positions as well as in predicting landing site.

To achieve the required alignment without violating biological quarantine barriers, an end-to-end calibration using reference mirror surfaces and collimated light beams after planetary vehicle assembly is suggested.

Sterilization -- The capsule is sterilized and enclosed in a sealed canister. From the time this is accomplished, the following interfaces will exist:

1) A physically sterile interface must be maintained between the canister and the spacecraft which is not violated by any physical connections between the spacecraft and capsule.

2) An interface exists between the capsule and spacecraft to ensure that canister jettisoning is accomplished as required to prevent contamination by spacecraft effluent or by contact with the external surfaces of the canister.

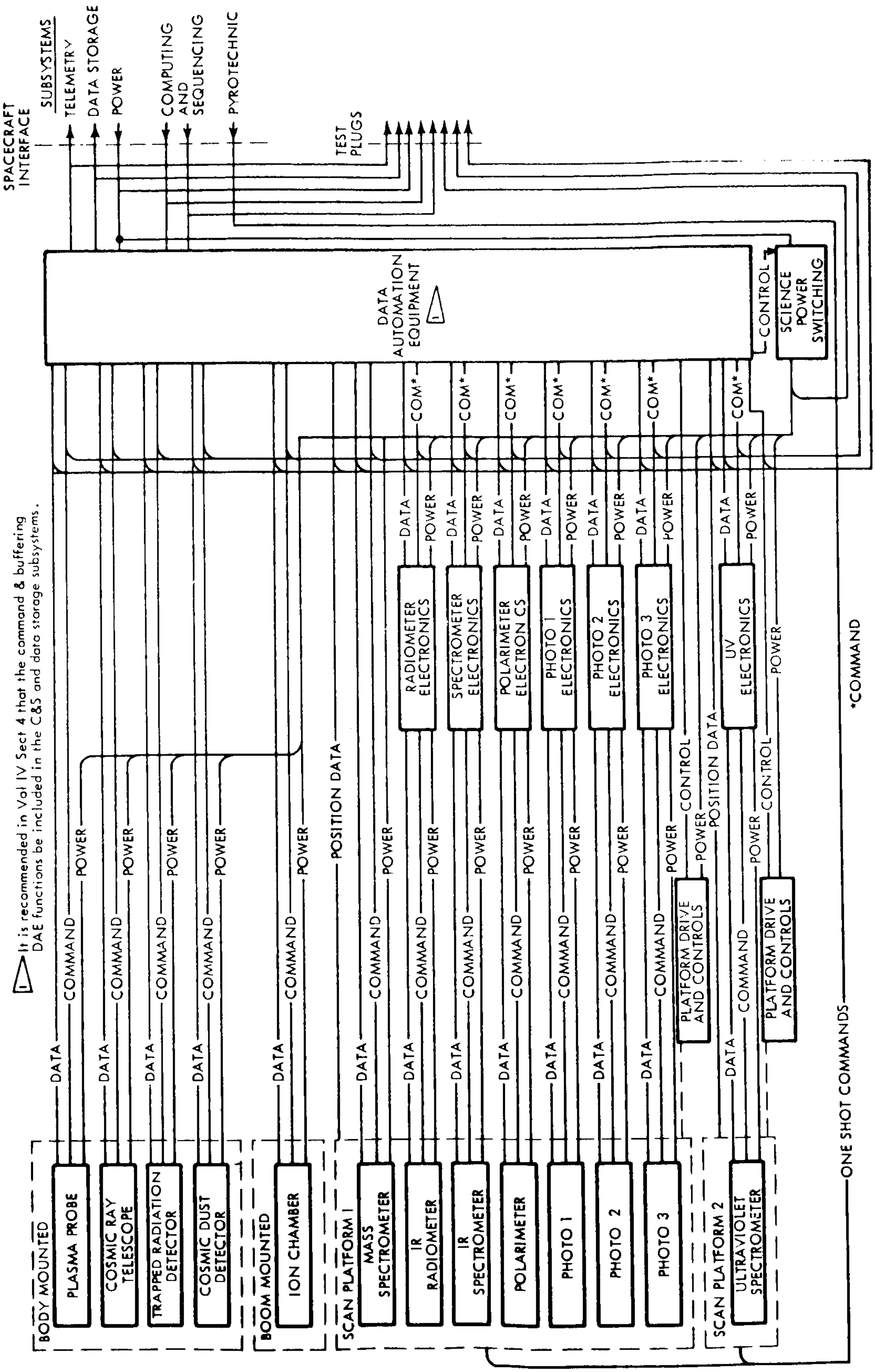


Figure 1.1.3-8: ELECTRICAL INTERFACES SPACECRAFT/SCIENCE SYSTEM

1.1.3.3 Spacecraft-Science Subsystem Interfaces

The science subsystem (1973 payload) consists of 12 scientific instruments designed to measure certain properties of Mars and its environment and the interplanetary medium between Earth and Mars. In addition to the instruments, the science subsystem includes auxiliary equipment such as the scan platforms, the power switching electronics, and the data automation equipment (DAE). The impact of the science subsystem on the spacecraft in terms of weight, power drain, and average data rate generated is shown in Table 1.1.3-1.

The structural and mechanical interfaces relate to the position and method of mounting the instruments on the spacecraft, as indicated below:

Instruments Mounted on Scan Platform No. 1:

1) Vidicons (three sensors)

2) Broadband IR spectrometer

3) High resolution IR spectrometer

4) IR radiometer

5) Atmospheric polarimeter

6) Atmospheric mass spectrometer

Mounted on Scan Platform No. 2:

7) UV spectrometer (This instrument requires a separate scan platform because of different pointing angle requirements.)

Mounted on one of the Low Gain Antenna Booms:

8) Ion chamber

Spacecraft Body-Mounted Instruments:

9) Plasma probe

10) Cosmic ray telescope

11) Cosmic dust detector

12) Trapped radiation detector.

All instruments must be mounted so that they satisfy their individual view-angle requirements.

Electronics associated with the science instruments are located within the electronics compartment in the spacecraft equipment module.

Table 1.1.3-1 : SCIENCE PAYLOAD EXPERIMENTS AND CHARACTERISTICS (1973 MISSION)

Experiment	Weight (lb)	Power (watts)	Data* (bps)
Imaging	150	60	5×10^5
Broadband IR Spectrometer	25	5	1330/665
Hi Resolution IR Spectrometer	30	14	1000
IR Radiometer	20	6	2300
UV Spectrometer	30	12	4000
Plasma Probe	10	3	23
Cosmic Ray Telescope	8	3	46
Cosmic Dust Detector	5	1	1.6
Trapped Radiation Detector	3	1	15.5
Ion Chamber	3	1	15.5
Atmospheric Polarimeter	9	3	300
Atmospheric Mass Spectrometer	8	10	60
Earth Occulation	0	0	0
Celestial Mechanics	0	0	0
Subtotal, Science Instruments	301	119	
DAE & Miscellaneous	70	44	
Total	371	163	

* Average during data acquisition period

Electrical interfaces include the power, computing and sequencing, telemetry, and data-storage subsystems. These interfaces are indicated in Figure 1.1.3-8.

Interfaces exist between the science subsystem and the temperature control subsystem because of the need to maintain certain instruments, and especially the electronics, within operating temperature design limits.

Pyrotechnics interfaces exist in the deployment of the scan platform and in lens-cover removal.

1.1.3.4 Spacecraft-Mission Operations System Interface

The mission operations system (MOS) includes the MOS-furnished mission-dependent equipment at the DSN sites, KSC, and AFETR; the computer programs for processing mission data and commands; the operations teams in the SFOF which conduct the mission operations from injection to the end of Mars orbit and Mars landed operations; and all other DSN, AFETR, and KSC facilities assigned to support the MOS in conduct of operations.

The spacecraft and the mission operation system have only functional interfaces. Operations teams interface with the spacecraft through the origination of commands and the analysis of telemetry data. The commands that can be sent by the operations team are listed in Table 2-2 of D2-115002-4. The telemetry data available for analysis is listed in Section 1.1.4 of this document.

MOS computer programs have functional interfaces with the spacecraft in the following areas:

1) Telemetry and Command Data Handling -- This will consist of an integrated set of computer programs written for both the DSIF and the SFOF. The primary functions of these programs are to process the incoming telemetry stream originating at the spacecraft, and process the command signals for transmission to the spacecraft.

2) Flight Path Analysis and Command -- This provides predicted positions, velocities, and maneuver sequences for the spacecraft trajectories.

3) Spacecraft Performance Analysis and Command -- This provides monitoring and control of the spacecraft subsystems. See Section 2.4.5 of D2-115002-4 for further details.

1.1.3.5 Spacecraft-Tracking and Data Acquisition System Interfaces

The tracking and data acquisition system (TDAS) includes the Deep Space Net and all other NASA and Department of Defense tracking and data acquisition stations, ships, and aircraft assigned to support the mission, all NASCOM and other circuits assigned to handle mission data and commands, and all other tracking and data acquisition personnel, physical facilities, and general support equipment assigned to handle mission data and commands and support mission operations.

The spacecraft and the data acquisition system have two basic interfaces: the S-band uplink and downlink. The uplink exists between the deep space instrumentation facilities' 85-foot antennas (early mission) and 210-foot antennas (late mission) and either of the spacecraft's two low gain antennas. The downlink signal originates at any one of the four spacecraft antennas and is received by one of the two DSIF antennas mentioned above. The uplink contains command, ranging, and doppler tracking information. The downlink contains telemetry data, scientific data, doppler tracking, angle tracking, and ranging information. The signal characteristics of these two links are discussed in Section 1.2.3.

1.1.4 Telemetry Channel List

A summary of the number and type of telemetry measurements for each spacecraft subsystem is shown in Table 1.1.4-1.

A detailed list of parameters measured for each subsystem is shown in Table 1.1.4-2. (The development of this measurements list is discussed in D2-115002-4, Section 2.0)

1.1.5 Guidance and Navigation Maneuver Errors

1.1.5.1 Guidance and Navigation Maneuver Error Requirements

The guidance and navigation maneuver error requirements are established in Section 1.1.1.11 of this volume.

1.1.5.2 Pointing-Angle Accuracy

The contribution to pointing error of each significant instrument error was calculated. The error sources included Sun and Canopus-sensor null offset, dead-zone and dead-zone switching-amplifier null offset errors, gyro drifts and misalignment, quantization, convertor and scale factor errors, thrust-vector-control null errors, and center-of-gravity offset errors. In each case, the maximum value of the error source coefficient, as it appears in the error equation, was used. The results, given in Table 1.1.5-1, show that the pointing-accuracy requirement is met for all maneuvers of a nominal mission. No requirement was established for capsule-off operation.

1.1.5.3 Δ V Magnitude Accuracy

The ΔV magnitude accuracy for the midcourse and orbit-insertion maneuvers is given in Table 1.1.5-2. For both the midcourse corrections and insertion Δ V maneuvers, ΔV magnitude accuracy is controlled by accelerometer measurement accuracy.

1.1.6 Flight Sequence

The nominal sequence of operations performance by the flight spacecraft from the period immediately preceding launch until mission completion is summarized in Figure 1.1.6-1. The vertical columns on the correlation chart represent mission phases as contained in Performance and Design Requirements for the 1973 Voyager Mission, General Specification for, dated January 1, 1967. The horizontal rows of the chart represent the flight spacecraft subsystems and Voyager support systems and their required functions per flight phase.

Table 1.1.4-1: TELEMETRY MEASUREMENT SUMMARY

SUBSYSTEM	NUMBER OF MEASUREMENTS		
	ANALOG	DIGITAL	TOTAL
ANTENNA	3	5	8
STRUCTURAL & MECHANICAL	17	7	24
TEMPERATURE CONTROL	61	0	61
PYROTECHNICS	0	8	8
SCIENCE	24	9	33
RADIO	43	5	48
TELEMETRY	10	18	28
GUIDANCE AND CONTROL	36	34	70
DATA STORAGE	16	48	64
COMPUTING & SEQUENCING	1	54	55
POWER	39	1	40
PROPULSION	14	13	27
TOTAL	264	202	466

Table 1.1.4-2: SPACECRAFT TELEMETRY CHANNEL LIST
(Sheet 1 of 9)

MEASUREMENTS			SIGNAL RANGE
SOURCE	QTY.	UNITS	
ANTENNA			
High-Gain Antenna Deployment	1	position	1 bit
High-Gain Antenna Position	2	angle	0-5 v
Low-Gain Antenna Deployment	2	position	1 bit
Med.-Gain Antenna Position	1	angle	0-5 v
Capsule Relay Antenna Deployment	1	position	1 bit
Med. Gain Antenna Deployment	1	position	1 bit
STRUCTURAL AND MECHANICAL SUBSYSTEM			
Science Scan Platform Deployment	5	position	1 bit
Science Scan Platform Position	3	angle	0-5 v
UV Platform Position	2	angle	0-5 v
Capsule Separation	1	position	1 bit
Accelerometer	6	ft/sec^2	0-5 v
Shroud Void Pressure	3	psia	0-5 v
PV Separation	1	position	1 bit
Shroud Void Temperature	3	°F	0-5 v
TEMPERATURE CONTROL SUBSYSTEM			
Louver Position	21	angle	0-5 v
Radiator Plate Temperature	21	°F	0-5 v
Solar Shield Temperature	2	°F	0-5 v
Engine Heat Shield Temperature	2	°F	0-5 v
Solar Panel Temperature	4	°F	0-5 v
Spacecraft Skin Temperature	2	°F	0-5 v
Antenna Temperature	3	°F	0-5 v
Propulsion Subsystem Temperature	4	°F	0-5 v
Science Scan Platform Temperature	2	°F	0-5 v
PYROTECHNICS SUBSYSTEM			
Pyrotechnic Safe Arm	2	condition(cond)	1 bit
Pyrotechnic Event Signals	4	event	4 bit
Pyrotechnic Power Status	2	cond	2 bit

Table 1.1.4-2: SPACECRAFT TELEMETRY CHANNEL LIST
(Sheet 2 of 9)

MEASUREMENT			SIGNAL RANGE
SOURCE	QTY.	UNITS	
SCIENCE SUBSYSTEM			
Plasma Probe Temperature	2	°F	0-5 v
Cosmic Ray Telescope Temperature	2	°F	0-5 v
Cosmic Dust Detector Temperature	2	°F	0-5 v
Trapped Radiation Detector Temperature	2	°F	0-5 v
Ion Chamber Temperature	2	°F	0-5 v
Ultraviolet Spectrometer Temperature	2	°F	0-5 v
High Resolution Infrared Spectrometer Temperature	2	°F	0-5 v
Photoimaging Temperature	2	°F	0-5 v
Infrared Scanner Temperature	2	°F	0-5 v
Broad Band IR Spectrometer Temperature	2	°F	0-5 v
Data Automation Equipment	6	status	1 bit
Data Automation Equipment Temperature	1	°F	0-5 v
Power Switching Electronics Temperature	1	°F	0-5 v
Power Switching Electronics	1	status	1 bit
Scan Platform No. 1 Temperature	1	°F	0-5 v
Scan Platform No. 1	1	status	1 bit
Scan Platform No. 2	1	status	1 bit
Scan Platform No. 2, Temperature	1	°F	0-5 v

Table 1.1.4-2: SPACECRAFT TELEMETRY CHANNEL LIST
(Sheet 3 of 9)

MEASUREMENT SOURCE	QTY.	UNITS	SIGNAL RANGE
RADIO SUBSYSTEM			
S-Band Receiver AGC (Coarse)	2	volts	0-5 v
S-Band Receiver AGC (Fine)	2	volts	0-5 v
S-Band Receiver Static Phase Error	2	volts	0-5 v
S-Band Receiver VCO Output Level	2	milliwatts	0-5v
S-Band Receiver LO Drive Level	2	milliwatts	0-5v
S-Band Receiver Regulated Voltage Level	2	volts	0-5 v
S-Band Receiver Temperature	2	°F	0-5 v
Exciter RF Power Output	2	milliwatts	0-5 v
Exciter Regulated Voltage Level	2	volts	0-5 v
Exciter Temperature	2	°F	0-5 v
Power Amplifier RF Power Output	2	watts	0-5 v
Power Amplifier Anode Voltage	2	volts	0-5v
Power Amplifier Helix Current	2	milliamps	0-5v
Power Amplifier Collector Current	2	milliamps	0-5v
Power Amplifier Collector Temperature	2	°F	0-5 v
Power Amplifier Converter Temperature	2	°F	0-5 v
Launch Transmitter Temperature	1	°F	0-5v

Table 1.1.4-2: SPACECRAFT TELEMETRY CHANNEL LIST
(Sheet 4 of 9)

MEASUREMENT SOURCE	QTY.	UNITS	SIGNAL RANGE
RADIO SUBSYSTEM (CONT)			
Relay Receiver AGC	2	volts	0-5v
Relay Receiver Performance	4	volts	0-5v
Relay Receiver Temperature	2	°F	0-5v
Relay Receiver Detector Lock	2	cond	1 bit
VCO Temperature	2	°F	0-5v
Power Sources A or B	1	cond	1 bit
VCO Counter	2	count	10 bits
TELEMETRY SUBSYSTEM			
Vehicle Identification	1	vehicle	2 bits
Temperature	2	°F	0-5v
Reference Voltages	4	volts	0-5v
Power Supply Voltages	4	volts	0-5v
Power Supply Status	8	cond	1 bit
Oscillator Status	1	cond	1 bit
T/M Mode	6	cond	1 bit
Frame Syne	1	sync.	15 bit
Frame Time	1	time	12 bit
GUIDANCE AND CONTROL SUBSYSTEM			
Star Mapping Signal No. 1	1	stellar magnitude	0-5v
Star Mapping Signal No. 2	1	stellar magnitude	0-5v

Table 1.1.4-2: SPACECRAFT TELEMETRY CHANNEL LIST
(Sheet 5 of 9)

MEASUREMENT			SIGNAL RANGE
SOURCE	QTY.	UNITS	
GUIDANCE AND CONTROL SUBSYSTEM (CONT)			
Canopus Roll Error No. 1	1	angle	0-5v
Canopus Roll Error No. 2	1	angle	0-5v
Canopus Recognition No. 1	1	cond	1 bit
Canopus Recognition No. 2	1	cond	1 bit
Limb and Terminator Sensor	2	cond	1 bit
IRU Roll Position	2	angle	0-5v
IRU Pitch Position	2	angle	0-5v
IRU Yaw Position	2	angle	0-5v
IRU Roll Rate	2	angular rate	0-5v
IRU Pitch Rate	2	angular rate	0-5v
IRU Yaw Rate	2	angular rate	0-5v
Roll Spin Motor No. 1	1	sync cond	1 bit
Pitch Spin Motor No. 1	1	sync cond	1 bit
Yaw Spin Motor No. 1	1	sync cond	1 bit
Roll Spin Motor No. 2	1	sync cond	1 bit
Pitch Spin Motor No. 2	1	sync cond	1 bit
Yaw spin Motor No. 2	1	sync cond	1 bit

Table 1.1.4-2: SPACECRAFT TELEMETRY CHANNEL LIST
(Sheet 6 of 9)

MEASUREMENT			SIGNAL RANGE
SOURCE	QTY.	UNITS	
GUIDANCE AND CONTROL SUBSYSTEM (CONT)			
Roll Failure Detect	2	cond	1 bit
Pitch Failure Detect	2	cond	1 bit
Yaw Failure Detect	2	cond	1 bit
Roll Gyro No. 1, Temperature	1	°F	0 - 5v
Pitch Gyro No. 1, Temperature	1	°F	0 - 5v
Yaw Gyro No. 1, Temperature	1	°F	0 - 5v
Roll Gyro No. 2, Temperature	1	°F	0 - 5v
Pitch Gyro No. 2, Temperature	1	°F	0 - 5v
Yaw Gyro No. 2, Temperature	1	°F	0 - 5v
Sun Acquisition Signal No. 1	1	cond	1 bit
Sun Acquisition Signal No. 2	1	cond	1 bit
Sun Sensor A Pitch Error	1	angle	0 - 5v
Sun Sensor A Yaw Error	1	angle	0 - 5v
Sun Sensor B Pitch Error	1	angle	0 - 5v
Sun Sensor B Yaw Error	1	angle	0 - 5v
Nitrogen Pressure, Manifold	2	psia	0 - 5v
Nitrogen Temperature	4	°F	0 - 5v
TVC Pitch Position	1	angle	0 - 5v
TVC Yaw Position	1	angle	0 - 5v
TVC Actuator Temperature	2	°F	0 - 5v
Thruster Voltage On/Off	16	cond	1 bit

Table 1.1.4-2: SPACECRAFT TELEMETRY CHANNEL LIST
(Sheet 7 of 9)

MEASUREMENT			SIGNAL RANGE
SOURCE	QTY.	UNITS	
DATA STORAGE SUBSYSTEM			
Recorder Case Temperature	8	°F	0-5v
Recorder Pressure	8	psia	0-5v
Malfunction	8	cond	1 bit
Tape Content	8	inches	7 bits
Record On/Off	8	cond	1 bit
Tape Speed Mode	8	cond	1 bit
End of Tape	8	cond	1 bit
Start of Tape	8	cond	1 bit
COMPUTING & SEQUENCING SUBSYSTEM			
Programmer Data	10	words	27 bits
C&S Ready	1	cond	1 bit
Parity Errors	6	cond	1 bit
Propulsion Rectifier Voltage	1	volts	0-5v
Accelerometer Null Detector	2	cond	1 bit
S/C Time	1	sec	27 bits
Status Signals	31	cond	1 bit
Command Word A	1	cond	1 bit
Command Word B	1	cond	1 bit
Command Status	1	word	27 bits
POWER SUBSYSTEM			
Battery Voltage	3	volts	0-5v
Maneuver Bus Voltage	1	volts	0-5v

Table 1.1.4-2: SPACECRAFT TELEMETRY CHANNEL LIST
(Sheet 8 of 9)

MEASUREMENT			SIGNAL RANGE
SOURCE	QTY.	UNITS	
POWER SUBSYSTEM (CONT)			
Solar Array Current	1	amps	0-5v
S/C Current	1	amps	0-5v
Unregulated DC Bus Voltage	1	volts	0-5v
Maneuver Bus Voltage	1	volts	0-5v
2400 Hz Inverter Output Voltage	3	volts	0-5v
2400 Hz Inverter Temperature	3	°F	0-5v
2400 Hz Inverter Output Current	3	amps	0-5v
2400 Hz Inverter Output Frequency	3	Hz	0-5v
Battery Charger Output Current	3	amps	0-5v
Battery Charger Temperature	3	°F	0-5v
Solar Panel Temperature	4	°F	0-5v
2400 Hz Inverter Temperature	3	°F	0-5v
Batteries A, B and C Temperature	3	°F	0-5v
Capsule DC Current	2	amps	0-5v
CapsuleVoltage	1	volts	0-5v
Solar Gate	1	cond	1 bit
PROPULSION SUBSYSTEM			
Solenoid Valves (engine low thrust)	8	position	1 bit
Solenoid Valves (engine high thrust)	4	position	1 bit
Helium Pressure Transducer	1	psia	0-5v
Helium Temperature Transducer	2	°F	0-5v

Table 1.1.4-2: SPACECRAFT TELEMETRY CHANNEL LIST
(Sheet 9 of 9)

MEASUREMENT			SIGNAL RANGE
SOURCE	QTY.	UNITS	
PROPULSION TELEMETRY SUBSYSTEM (CONT)			
Fuel Pressure Transducer	1	psia	0-5v
Fuel Temperature Transducer	2	°F	0-5v
Engine Chamber Pressure Transducer	1	psia	0-5v
Oxidizer Pressure Transducer	1	psia	0-5v
Oxidizer Temperature Transducer	2	°F	0-5v
Oxidizer Start Tank Flow Meter	1	lb/sec	0-5v
Fuel Start Tank Flow Meter	1	lb/sec	0-5v
Oxidizer Main Tank Flow Meter	1	lb/sec	0-5v
Fuel Main Tank Flow Meter	1	lb/sec	0-5v
Throttle Actuator Position	1	position	1 bit

Table 1.1.5-1: POINTING ERROR ANALYSIS

ERROR SOURCE	UNITS	ERROR (3σ)	PITCH MULTIPLIER	MAX. VALUE	YAW MULTIPLIER	MAX. VALUE	SQUARED ERROR
MANEUVER ERROR							
Roll Drift Rate	r/s	$1.458\text{X}10^{-6}$	$\left[\frac{1}{\dot\theta_y}(1-\cos\theta_y) - \sin\theta_y\left(\frac{\theta_r}{\dot\theta_r}+t_1\right)\right]$	571	0		6.93×10^{-7}
Pitch Drift Rate	r/s	1.458×10^{-6}	$\left[\frac{-\sin\theta_y}{\dot\theta_y}+t_2\right]$	346	0		2.54×10^{-7}
Yaw Drift Rate	r/s	1.458×10^{-6}	0		$\left[\frac{\theta_y}{\dot\theta_y}+t_2\right]$	960	1.96×10^{-6}
Roll Gyro Misalignment with Pitch Axis	rad	8.93×10^{-4}	$-\sin\theta_r\cos\theta_y$	1	$(1-\cos\theta_r)$	2	3.99×10^{-6}
Roll Gyro Misalignment with Yaw Axis	rad	8.93×10^{-4}	$\cos\theta_y(1-\cos\theta_r)$	2	$\sin\theta_r$	1	3.99×10^{-6}
Pitch Gyro Misalignment with Roll Axis	rad	---	0	0	0	0	---
Pitch Gyro Misalignment with Yaw Axis	rad	8.93×10^{-4}	$\sin\theta_y$	1	0	0	7.99×10^{-7}
Yaw Gyro Misalignment with Pitch Axis	rad	8.93×10^{-4}	$\sin\theta_y$	1	0	0	7.99×10^{-7}
Yaw Gyro Misalignment with Roll Axis	rad	8.93×10^{-4}	$(1-\cos\theta_y)$	2	0	0	3.18×10^{-6}
Roll Limit Cycle Magnitude	rad	9.05×10^{-3}	$-\sin\theta_y$	1	0	0	81.5×10^{-6}
Roll Sensor Accuracy	rad	5.3×10^{-3}	$-\sin\theta_y$	1	0	0	28.1×10^{-6}
Roll Switching Amplifier Null Offset	rad	5.23×10^{-4}	$-\sin\theta_y$	1	0	0	2.73×10^{-7}
Pitch Limit Cycle Magnitude	rad	9.05×10^{-3}	$\cos\theta_y$	1	0	0	81.5×10^{-6}
Pitch Sensor Accuracy	rad	2.77×10^{-3}	$\cos\theta_y$	1	0	0	7.65×10^{-6}
Pitch Switching Amplifier Null Offset	rad	5.23×10^{-4}	$\cos\theta_y$	1	0	0	2.73×10^{-7}
Yaw Limit Cycle Magnitude	rad	9.05×10^{-3}	0	0	1	1	81.5×10^{-6}
Yaw Sensor Accuracy	rad	2.77×10^{-3}	0	0	1	1	7.65×10^{-6}
Yaw Switching Amplifier Null Offset	rad	5.23×10^{-4}	0	0	1	1	2.73×10^{-7}
Roll Quantization Error	rad	1.745×10^{-4}	$-\sin\theta_y$	1	0	0	3.05×10^{-8}
Roll V/F Converter Error	r/s	1.05×10^{-5}	$-\frac{\theta_r}{\dot\theta_r}\sin y$	225	0	0	5.57×10^{-6}
Roll Gyro Scale Factor Error	rad	3.92×10^{-4}	$-\sin\theta_y$	1	0	0	1.54×10^{-7}
Yaw Quantization Error	rad	1.74×10^{-4}	0	0	1	1	3.05×10^{-8}
Yaw V/F Converter Error	r/s	1.05×10^{-5}	0	0	$\frac{\theta_y}{\dot\theta_y}$	900	89.3×10^{-6}
Yaw Gyro Scale Factor	rad	1.57×10^{-3}	0	0	1	1	2.46×10^{-6}
							452.06×10^{-5}
					RSS Maneuver Error =		21.26 mrad (1.22 deg)

Thrust Vector Control Errors		Pitch Error	Yaw Error	Squared Error
Midcourse	rad	1.74×10^{-2}	1.74×10^{-2}	6.05×10^{-4}
Orbit Insertion Capsule Off	rad	3.84×10^{-2}	3.84×10^{-2}	29.49×10^{-4}
Orbit Insertion Capsule On	rad	1.74×10^{-2}	1.74×10^{-2}	6.05×10^{-4}
Orbit Trim Capsule Off	rad	2.79×10^{-2}	2.79×10^{-2}	15.57×10^{-4}
Orbit Trim Capsule On	rad	1.74×10^{-2}	1.74×10^{-2}	6.05×10^{-4}

TOTAL POINTING ERRORS (3)	UNITS	CAPSULE ON	CAPSULE OFF
Midcourse	m rad	32.5 (1.86°)	
Orbit Insertion	m rad	32.5 (1.86°)	58.3 (3.39°)
Orbit Trim	m rad	32.5 (1.86°)	44.8 (2.56°)

Table 1.1.5-2: ΔV MAGNITUDE ERROR TABULATION

MIDCOURSE CORRECTION		
Error Source	Magnitude Error, 3σ	ΔV Error, 3σ Meters/Sec
Accelerometer Null Bias	2×10^{-4}g	0.42%, ΔV
Accelerometer Scale Factor	0.1%	0.1%, ΔV
Accelerometer Resolution	0.02 m/sec	0.02 m/sec
Engine Tailoff Uncertainty	60 lb-sec	0.03 m/sec
Total RSS Error = $\left[(0.0043\,\Delta V)^2 + 0.036^2\right]^{1/2}$ m/sec, 3σ		
ORBIT INSERTION		
Error Source	Magnitude Error, 3σ	ΔV Error, 3σ
Accelerometer Null Bias	2×10^{-4}g	0.6 m/sec
Accelerometer Scale Factor	0.1%	1.25 m/sec
Accelerometer Resolution	0.02 m/sec	0.02 m/sec
Engine Tailoff Uncertainty	143 lb-sec	0.11 m/sec
RSS Total = 1.4 m/sec 3σ		

The comparison chart represents a nominal flight for either of the two spacecraft, since they are identical in design and functional performance. Accomplishment of the subsystem operations is initiated and sequenced by a programmed computer and sequencer. Any event or sequence of events within the described flight sequence can be altered by commands from Earth.

1.1.7 Spacecraft Layout and Configuration

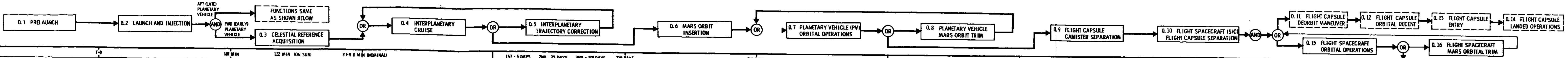
The baseline configuration is shown in Figure 1.1.7-1.

In addition to the design requirements, a number of design objectives were established to ensure a near optimum configuration. Major objectives met in the development of the design were:

1) High packaging density of subsystems and equipment to minimize volume

2) Modularization of certain subsystems

3) Efficient structural concepts

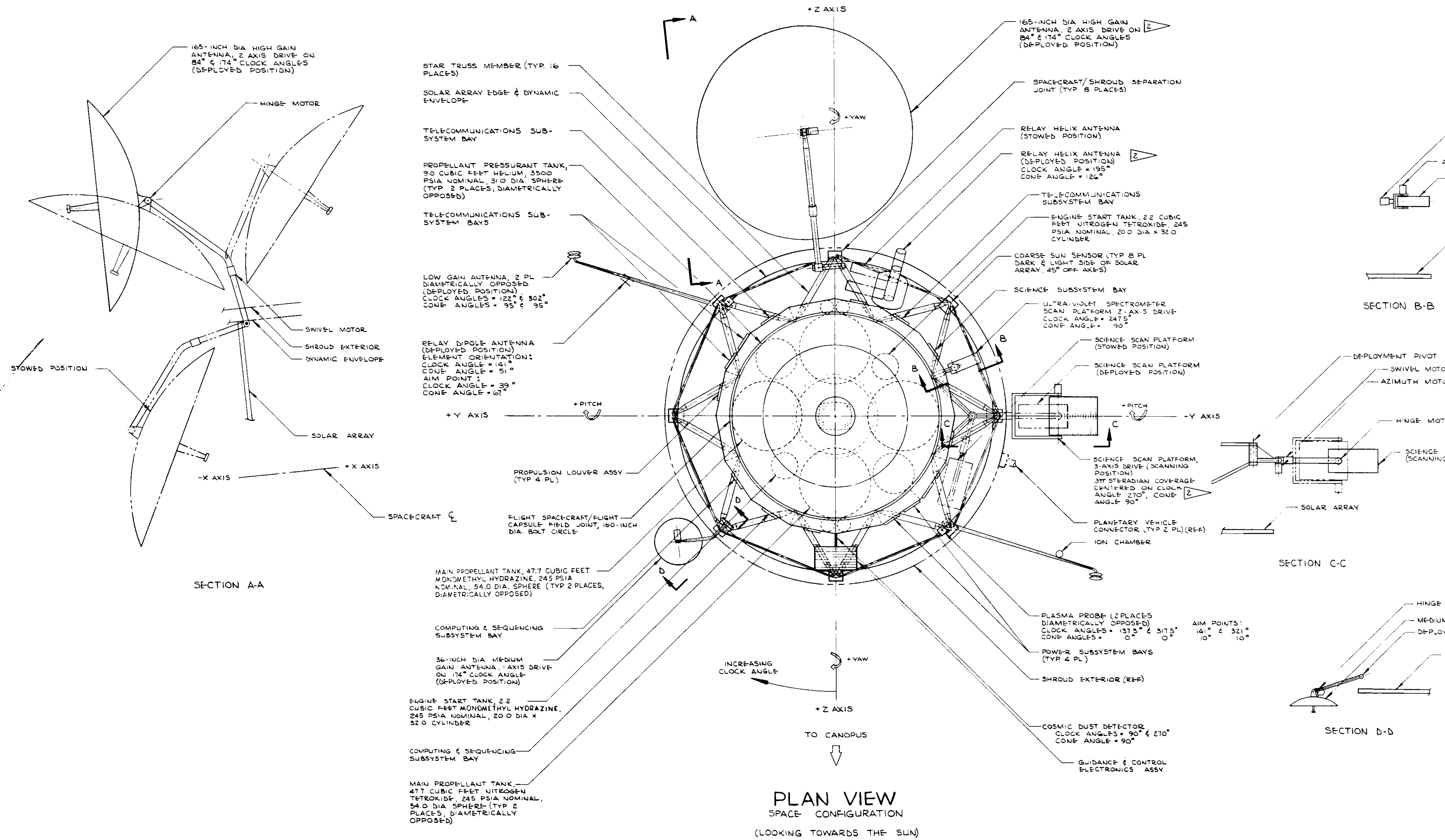


	0.1 PRELAUNCH	0.2 LAUNCH AND INJECTION	0.3 CELESTIAL REFERENCE ACQUISITION	0.4 INTERPLANETARY CRUISE
TIME: FWD PLANETARY VEHICLE		T-0	109 MIN 122 MIN (ON SUN)	8 HR 0 MIN (NOMINAL)
TIME: AFT PLANETARY VEHICLE		T-0	123 MIN 137 MIN (ON SUN)	8 HR 15 MIN (NOMINAL)
LOS MOS DSN	A MONITOR S/C CONDITIONED AIR B CONDUCT S/C & LV CHECKS C CHECK RANGE SAFETY D RF & TELEMETRY CHECK E EVALUATE TELEMETRY AT DSIF 71 F VERIFY PV SYSTEMS G VERIFY PV OPERATION ON EXTERNAL POWER H VERIFY PV OPERATION ON INTERNAL POWER I PERFORM LCE SYSTEMS TESTS J PERFORM AUTOMATIC COUNTDOWN K S COMMAND FIRST-STAGE IGNITION L COMMAND SWITCHOVER TO S/C INTERNAL POWER M COMMAND START C&S CLOCK N COMMAND UPDATE S/C C&S MEMORY	RANGE A MONITOR BOOSTER SEPARATION B TRACK DURING BOOST C RELAY TRACKING DATA TO LOS D PROVIDE BOOSTER COMMANDS DSIF 71 A STANDBY ON ALERT B RECEIVE ANTENNA SEARCH DATA C SEARCH & ACQUIRE SPACE VEHICLE D ESTABLISH & VERIFY CONTROL E TRACK SPACECRAFT SFOF/MOS A STANDBY ON ALERT B ASSUME CONTROL AFTER INJECTION INTO HELIOCENTRIC ORBIT C ANALYZE DATA & UPDATE TRAJECTORY PARAMETERS D GENERATE & TRANSMIT DSIF TRACKING PREDICTION DATA DSN A RECEIVE DATA FROM DSIF 71 B STANDBY C TRACK	DSN A MONITOR PV DOWNLINK TRANSMISSION B ACHIEVE DOWNLINK/UPLINK PHASE LOCK C MONITOR PV DEPLOYMENT •ANTENNAS D RECEIVE SOLAR ACQUISITION DATA & RELAY E RECEIVE & RELAY ROLL CALIBRATION DATA TO SFOF F RECEIVE & RELAY CANOPUS ACQUISITION DATA TO SFOF G RELAY BACKUP COMMAND REACQUIRE CANOPUS AS REQUIRED SFOF A RECEIVE MONITOR DATA FROM DSN B RECEIVE & ANALYZE ROLL CALIBRATION DATA C RECEIVE & ANALYZE CANOPUS ACQUISITION DATA D VERIFY CANOPUS ACQUISITION OR COMMAND CANOPUS REACQUISITION	DSN A TRACK PV B RECEIVE ENGINEERING & SCIENCE DATA & RELAY TO SFOF C PROVIDE PV RANGE DATA (DERIVED) D COMMAND DATA PLAYBACK E SWITCH FROM 85-FT NET TO 210-FT NET SFOF A RECEIVE ENGINEERING & SCIENCE DATA & DISPLAY B RECEIVE & ANALYZE TRACK & RANGE DATA C COMPUTE TRAJECTORY PARAMETERS
POWER	A SWITCH S/C LOADS FROM EXTERNAL POWER TO BATTERY	A PROVIDE BATTERY POWER TO S/C SUBSYSTEMS	A TRANSFER FROM BATTERY TO SOLAR PANEL MODE B TURN ON BATTERY CHARGER C PROVIDE POWER TO S/C & FLIGHT CAPSULE	A PROVIDE SOLAR POWER TO S/C SUBSYSTEMS B SWITCH ON CAPSULE POWER AFTER BATTERY CHARGE CONTROL TRANSFERS FROM FULL CHARGE TO TRICKLE CHARGE C PROVIDE TRICKLE CHARGE TO BATTERIES
COMPUTING & SEQUENCE (C&S)	A START SPACECRAFT (CUMULATIVE TIME REF) at T - 180 SEC B RECEIVE "UPDATE" GROUND COMMANDS	A RECEIVE SEPARATION SIGNAL FROM MECHANISMS	A COMMAND PYROTECHNICS UNLOCK & RELEASE ANTENNAS B COMMAND PYROTECHNICS BACKUP COMMANDS (AS REQUIRED) C COMMAND MANEUVER RECORDER "ON" D COMMAND 2 SCIENCE "ON" E COMMAND SOLAR ACQUISITION F COMMAND COMMUNICATION PHASE LOCK G COMMAND CALIBRATION ROLL H COMMAND CELESTIAL REFERENCE "ON" I COMMAND CAPSULE POWER "ON" J COMMAND DECODER BACKUP OF C&S (AS REQUIRED) K COMMAND PYROTECHNICS FIRE N_2 SQUIBS	A COMMAND CANOPUS CONE ANGLE UPDATE B COMMAND HIGH-GAIN ANTENNA ELEVATION & AZIMUTH C COMMAND RADIO SWITCH TO HIGH-GAIN ANTENNA D COMMAND RADIO SWITCH TO HIGH-POWER TRANSMITTER (T + 3 DAYS) E COMMAND RANGING "ON" & "OFF" F COMMAND HIGH-GAIN ANTENNA POSITION UPDATE G COMMAND GYROS "OFF" H COMMAND CAPSULE POWER "ON" I COMMAND DECODER BACKUP OF C&S (AS REQUIRED) J COMMAND PYROTECHNICS UNLOCK, UNCOVER, & RELEASE SCIENCE ELEMENTS
GUIDANCE AND CONTROL	A TURN ON G&C POWER	A G&C POWER "ON"	A GYROS IN RATE MODE, DAMP OUT PV TUMBLING RATE B PITCH & YAW PV, ACQUIRE SUN, SWITCH GYRO RATE "OUT" C STABILIZE PV DURING PHASE LOCK ACQUISITION •DERIVED RATE, PITCH & YAW AXES (SUN SENSOR) •ROLE RATE < .05°/SEC. D RELAY STAR MAP E ACQUIRE CANOPUS F SWITCH TO CANOPUS REF., GYRO RATE "OUT"	A GYROS "OFF" B UPDATE CANOPUS CONE ANGLE
RADIO	A RECEIVER "ON" B TRANSMIT SYSTEM STATUS TO LOS C LAUNCH TRANSMITTER "ON"	A RELAY DOWNLINK SIGNAL TO LAUNCH VEHICLE B RELAY ENGINEERING DATA TO LAUNCH VEHICLE C TRACK AND DEMODULATE UPLINK SIGNALS	A TRANSMIT DOWN LINK SIGNAL B TRANSMIT VERIFICATION DATA -- SOLAR ACQUISITION C TRANSMIT ROLL CALIBRATION DATA D TRANSMIT CANOPUS ACQUISITION DATA TO DSN E RECEIVE GROUND BACKUP COMMANDS & RELAY F ACHIEVE PHASE LOCK LOOP	A TRANSMIT ENGINEERING & SCIENCE DATA TO DSN B TRANSMIT PLAYBACK DATA (ON COMMAND FROM DSN) C SWITCH ON TWTA (AT 3 DAYS) D RECEIVE COMMANDS FROM DSN & RELAY TO C&S E MAINTAIN DOWNLINK PHASE LOCK
TELEMETRY	A SELECT MODE 1 B ENGINEERING DATA (BOTH S/C & CAPSULE) RELAY TO LOS VIA HARDLINE AND/OR RF LINK	A RELAY DATA TO BOOSTER TELEMETRY DURING POWERED FLIGHT B RECEIVE & ENCODE ENGINEERING DATA C RECEIVE & ENCODE SEPARATION SIGNALS	A TELEMETRY IN MODE 1 B RECEIVE DEPLOYMENT VERIFICATION SIGNALS C RECEIVE & ENCODE ROLL CALIBRATION DATA	A FORMAT S/C ENGINEERING & SCIENCE DATA B RECEIVE & FORMAT FLIGHT CAPSULE STATUS DATA
ANTENNA		A TRANSMIT VIA LOW-GAIN ANTENNA	A TRANSMIT AND RECEIVE VIA LOW GAIN ANTENNA	A SET HIGH-GAIN ANTENNA ELEVATION & AZIMUTH B SET MEDIUM-GAIN ANTENNA AZIMUTH C SWITCH TO HIGH-GAIN ANTENNA
DATA STORAGE		A ACTIVATE MANEUVER RECORDER AS LAUNCH TELEMETRY BACKUP	A STORE DATA DURING MANEUVER	A RECORD SCIENCE DATA (SOLAR FLARE MODE) B PLAYBACK RECORDED DATA
STRUCTURE & MECHANISMS	A PROVIDE STRUCTURAL SUPPORT TO S/C SUBSYSTEMS AND FLIGHT CAPSULE	A PROVIDE STRUCTURAL SUPPORT TO S/C SUBSYSTEMS AND CAPSULE B PROVIDE RELEASE & SEPARATION MECHANISM FOR FORWARD & AFT PV'S C PROVIDE SEPARATION SENSOR D PROVIDE SEPARATION SIGNALS TO TELEMETRY & C&S	A DEPLOY ANTENNAS B PROVIDE VERIFICATION SIGNALS TO TELEMETRY	A PROVIDE STRUCTURAL SUPPORT TO S/C SYSTEMS & CAPSULE B DEPLOY SCIENCE SCAN PLATFORM & INSTRUMENT COVERS -- TRANSMIT VERIFICATION SIGNAL TO TELEMETRY
PYROTECHNICS	———	A ARM PYROTECHNICS B FIRE SQUIBS, ACTIVATE SEPARATION THRUSTERS, FWD PV C FIRE SQUIBS, ACTIVATE SEPARATION THRUSTERS, AFT PV	A FIRE EED'S RELEASE ANTENNAS B FIRE REACTION CONTROL N_2 SQUIBS	A FIRE EED'S RELEASE SCIENCE SCAN PLATFORM AND COVERS
TEMPERATURE CONTROL	A MONITOR FLIGHT SPACECRAFT TEMPERATURES	A PROVIDE HEAT-SINK COOLING B MONITOR FLIGHT S/C TEMPERATURES	A PROVIDE HEAT-SINK COOLING B PROVIDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING C MONITOR FLIGHT S/C TEMPERATURES	A PROVIDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING B MONITOR FLIGHT SPACECRAFT TEMPERATURES
PROPULSION	———	———	———	———
SCIENCE	A ALL SCIENCE "OFF"	———	A CRUISE/ORBITAL SCIENCE "ON" B GATHER, ENCODE & RELAY MANEUVER MODE CRUISE DATA	A GATHER, ENCODE AND RELAY CRUISE/ORBITAL SCIENCE & SOLAR FLARE DATA B SWITCH TO SOLAR FLARE MODE DURING SOLAR FLARE
LAUNCH VEHICLE	A PROVIDE S/C ENVIRONMENT CONTROL (PRECHILL TO 40° ± 5° BY T - 1 HR) B AUTOMATIC COUNTDOWN C IGNITION S-IC ENGINES	A LAUNCH, S-1 & S-II BURN B S-IV FIRST BURN INSERT IN PARKING ORBIT C JETTISON NOSE SHROUD IN EARTH ORBIT D S-IVB SECOND BURN INSERT IN TRANS-MARS TRAJECTORY E PROVIDE SEPARATION SIGNAL TO FORWARD PV F MAINTAIN ATTITUDE CONTROL DURING FORWARD PV SEPARATION G JETTISON AFT SHROUD AFTER S-IVB BURN H PROVIDE SIGNAL TO SEPARATE AFT PV I MAINTAIN ATTITUDE CONTROL DURING AFT PV SEPARATION J APPLY RETROTHRUST		

	0.5 INTERPLANETARY TRAJECTORY CORRECTION	0.6 MARS ORBIT INSERTION	0.7 PLANETARY VEHICLE (PV) ORBITAL OPERATIONS	0.8 PLANETARY VEHICLE MARS ORBIT TRIM
TIME: FWD PLANETARY VEHICLE	1ST - 5 DAYS 2ND - 25 DAYS 3RD - 178 DAYS	219 DAYS	219 DAYS	
TIME: AFT PLANETARY VEHICLE	1ST - 5 DAYS 2ND - 25 DAYS 3RD - 186 DAYS	227 DAYS	227 DAYS	
LOS MOS DSN	DSN A TRANSMIT MANEUVER METERS TO PV •ROLL TURN MAGNITUDE & DIRECTION •YAW (OR PITCH) TURN MAGNITUDE & DIRECTION •MOTOR BURNTIME & ΔV B TRANSMIT INITIATE MANEUVER SEQUENCE COMMAND C RECEIVE VERIFICATION OF CORRECTION MANEUVER D TRACK PV E PROVIDE POINTING ANGLE & RANGE DATA F RELAY DATA TO SFOF G BACKUP CANOPUS ACQUISITION COMMAND SFOF A COMPUTE MANEUVER PARAMETERS & TRANSMIT TO DSN B VERIFY CORRECTION MANEUVER C RECEIVE ENGINEERING & SCIENCE DATA & DISPLAY	DSN A RELAY ORBIT INSERTION PARAMETERS B TRACK PV C TRANSMIT MANEUVER PARAMETERS •ROLL TURN MAGNITUDE & DIRECTION •YAW (OR PITCH) TURN MAGNITUDE & DIRECTION •MOTOR BURN TIME & ΔV D RECEIVE ENGINEERING & SCIENCE DATA & RELAY TO SFOF E PROVIDE PV RANGE DATA F COMMAND DATA PLAYBACK G RECEIVE VERIFICATION OF ORBIT INSERTION SEQUENCE & RELAY TO SFOF SFOF A COMPUTE ORBIT INSERTION PARAMETERS & PROVIDE BACKUP COMMANDS AS REQUIRED B RECEIVE ENGINEERING & SCIENCE DATA & DISPLAY C RECEIVE & ANALYZE TRACK & RANGE DATA D COMPUTE TRAJECTORY PARAMETERS E RECEIVE ORBIT INSERTION VERIFICATION	DSN A TRACK PV B RECEIVE ENGINEERING & SCIENCE DATA & RELAY TO SFOF C PROVIDE PV RANGE DATA D COMMAND DATA PLAYBACK E RECEIVE MARS ORBIT DATA; RELAY TO SFOF SFOF A RECEIVE ENGINEERING & SCIENCE DATA & DISPLAY B RECEIVE & ANALYZE TRACK & RANGE DATA C COMPUTE TRAJECTORY PARAMETERS D RECEIVE & DISPLAY MARS ORBIT DATA E COMPUTE ORBIT PARAMETERS	DSN A TRANSMIT MANEUVER PARAMETERS TO PV •ROLL TURN MAGNITUDE AND DIRECTION •YAW (OR PITCH) TURN MAGNITUDE & DIRECTION •MOTOR BURNTIME & ΔV B TRANSMIT INITIATE MANEUVER SEQUENCE COMMAND C RECEIVE VERIFICATION OF CORRECTION MANEUVER D TRACK PV E PROVIDE POINTING ANGLE & RANGE DATA TO ALL DSS SFOF A COMPUTE MANEUVER PARAMETERS & TRANSMIT TO DSN B VERIFY CORRECTION MANEUVER C RECEIVE & DISPLAY ENGINEERING & SCIENCE DATA
POWER	A SWITCH TO BATTERY POWER B TURN OFF POWER TO CAPSULE C SWITCH TO SOLAR POWER MODE AFTER MANEUVERS D START CHARGING BATTERIES	A SWITCH TO BATTERY B TURN OFF POWER TO CAPSULE C SWITCH TO SOLAR POWER MODE AFTER MANEUVER D START CHARGING BATTERIES	A PROVIDE SOLAR POWER TO S/C SUBSYSTEMS B SWITCH ON CAPSULE POWER AFTER BATTERY CHARGE CONTROL TRANSFERS FROM FULL CHARGE TO TRICKLE CHARGE C PROVIDE TRICKLE CHARGE TO BATTERIES	A SWITCH TO BATTERY (NO POWER TO CAPSULE DURING OFF-SUN PERIODS) B SWITCH TO SOLAR POWER MODE AFTER MANEUVER C START CHARGING BATTERIES
COMPUTING & SEQUENCE (C&S)	A RECEIVE MANEUVER PARAMETERS FROM COMMAND DECODER BACKUP (AS REQUIRED) B COMMAND MANEUVER RECORDER "ON" C COMMAND RADIO SWITCH TO LOW-GAIN ANTENNA D COMMAND ACCELEROMETER AND GYROS "ON" E COMMAND ATTITUDE CHANGE (ROLL, PITCH OR YAW) F COMMAND ENABLE TVC ACTUATOR AND ELECTRONICS G COMMAND PYROTECHNIC FIRE EED'S TO ACTIVATE SQUIB VALVES-PROP SYSTEM H COMMAND CORRECTION MANEUVER FIRING, COUNT ΔV & TERMINATE I COMMAND TVC "OFF" J COMMAND REVERSE MANEUVERS K COMMAND REACQUISITION OF CELESTIAL REF. L COMMAND SWITCH TO HIGH-GAIN ANTENNA M COMMAND MANEUVER RECORDER "OFF"; SCIENCE MODE "ON" N COMMAND DATA STORAGE PLAYBACK O COMMAND CAPSULE POWER "OFF" P COMMAND ACCELEROMETER & GYROS "OFF"	A RECEIVE MANEUVER PARAMETERS FROM COMMAND DECODER BACKUP (AS REQUIRED) B COMMAND MANEUVER RECORDER "ON" C COMMAND RADIO SWITCH TO MEDIUM-GAIN ANTENNA D COMMAND S/C ATTITUDE CHANGE E COMMAND ENABLE TVC ACTUATOR & ELECTRONICS F COMMAND PYROTECHNIC FIRE EED'S TO FIRE SQUIBS (PROPULSION) G COMMAND INSERTION MANEUVER FIRING, COUNT ΔV, AND TERMINATE H COMMAND TVC "OFF" I COMMAND ACCELEROMETER "OFF" J COMMAND REVERSE MANEUVERS K COMMAND REACQUISITION OF CELESTIAL REFERENCES (ROLL, PITCH OR YAW) L RECEIVE OPERATIONAL EVENT DATA FROM G&C M COMMAND SWITCH TO HIGH GAIN ANTENNA N COMMAND MANEUVER RECORDER "OFF"; SCIENCE MODE "ON" O COMMAND DATA STORAGE PLAYBACK P COMMAND ACCELEROMETER & GYROS "OFF"	A COMMAND LIMB/TERMINATOR UNIT "ON" B COMMAND ORBITAL SCIENCE "ON"; CRUISE "ON" C COMMAND CANOPUS CONE ANGLE UPDATE D COMMAND HIGH-GAIN ANTENNA POSITION UPDATE E COMMAND RANGING "ON" & "OFF" F COMMAND GYROS ON IF SUN OCCULTATIONS OR CANOPUS INTERFERENCE MAY OCCUR IN THIS ORBIT G COMMAND CAPSULE POWER "ON" H COMMAND DECODER BACKUP OF C&S (AS REQUIRED)	A RECEIVE MANEUVER PARAMETERS FROM COMMAND DECODER BACKUP (AS REQUIRED) B COMMAND MANEUVER RECORDER "ON" C COMMAND GYROS & ACCELEROMETER "ON" D COMMAND RADIO SWITCH TO MEDIUM-GAIN ANTENNA E COMMAND S/C ATTITUDE CHANGE F COMMAND ENABLE TVC ACTUATOR & ELECTRONICS G COMMAND INSERTION MANEUVER FIRING, COUNT ΔV, AND TERMINATE H COMMAND TVC "OFF" I COMMAND ACCELEROMETER "OFF" J COMMAND REVERSE MANEUVERS K COMMAND REACQUISITION OF CELESTIAL REFERENCES (ROLL, PITCH OR YAW) L RECEIVE OPERATIONAL EVENT DATA FROM G&C M COMMAND SWITCH TO HIGH-GAIN ANTENNA N COMMAND MANEUVER RECORDER "OFF"; SCIENCE MODE "ON" O COMMAND DATA STORAGE PLAYBACK P COMMAND ACCELEROMETER & GYROS "OFF"
GUIDANCE AND CONTROL	A GYROS AND ACCELEROMETER "ON" B POSITION S/C FOR MANEUVER (ROLL, PITCH OR YAW) C INERTIAL "HOLD" MODE FOR ΔV (CONTROL PITCH & YAW WITH TVC; ROLL BY JETS) D ENABLE TVC ACTUATOR & ELECTRONICS E PITCH OR YAW BACK, REACQUIRE SUN F ROLL BACK, REACQUIRE CANOPUS G ACCELEROMETER MEASURES ΔV INCREMENT H TVC AND ACCELEROMETER "OFF"	A GYROS AND ACCELEROMETER "ON" B POSITION S/C FOR MANEUVER (ROLL, PITCH OR YAW) C INERTIAL HOLD MODE FOR ΔV (CONTROL PITCH & YAW ATTITUDE WITH TVC, YAW BY REACTION JETS) D ENABLE TVC ACTUATOR & ELECTRONICS E YAW OR PITCH BACK TO PREVIOUS ATTITUDE F REACQUIRE SUN, SWITCH GYRO RATE "OUT" ROLL AXIS G ROLL BACK TO PREVIOUS ATTITUDE H REACQUIRE CANOPUS, SWITCH GYRO RATE "OUT" ROLL AXIS I ACCELEROMETER MEASURES ΔV INCREMENT J TVC AND ACCELEROMETER "OFF"	A SENSE CANOPUS OCCULTED; SWITCH ROLL AXIS TO INERTIAL "HOLD" B SENSE CANOPUS PRESENT, REACQUIRE CANOPUS; SWITCH ROLL GYRO TO "RATE" MODE C SENSE LIMB & TERMINATOR CROSSINGS, RELAY TO C&S D UPDATE CANOPUS CONE ANGLE E SENSE LOSS OF SUN PRESENCE, SWITCH PITCH & YAW AXES TO INERTIAL HOLD IF OCCULTATION SHOULD OCCUR F SENSE SUN PRESENCE, REACQUIRE SUN, SWITCH PITCH & YAW AXES TO "RATE" MODE	A GYROS AND ACCELEROMETER "ON" B POSITION S/C FOR CORRECTION MANEUVER (ROLL, PITCH OR YAW) C INERTIAL HOLD MODE FOR ΔV (CONTROL PITCH OR YAW ATTITUDE WITH TVC, YAW BY REACTION JETS) D ENABLE TVC ACTUATOR & ELECTRONICS E YAW OR PITCH BACK TO PREVIOUS ATTITUDE F REACQUIRE SUN, SWITCH GYRO RATE "OUT" ROLL AXIS G ROLL BACK TO PREVIOUS ATTITUDE H REACQUIRE CANOPUS, SWITCH GYRO RATE "OUT" ROLL AXIS I ACCELEROMETER MEASURES ΔV INCREMENT J TVC AND ACCELEROMETER "OFF"
RADIO	A RECEIVE MANEUVER PARAMETERS FROM DSN & RELAY TO C&S B RECEIVE INITIATE MANEUVER COMMAND & RELAY TO C&S C RELAY MANEUVER VERIFICATION TO DSN D TRANSMIT ENGR. & SCIENCE DATA TO DSN (MODE 1)	A RECEIVE INSERTION PARAMETERS FROM DSN (AS REQUIRED) & RELAY TO C&S B REESTABLISH PHASE LOCK AFTER INSERTION C TRANSMIT ENGINEERING DATA & SCIENCE DATA TO DSN D TRANSMIT ORBIT INSERTION VERIFICATION TO DSN	A MAINTAIN DOWNLINK PHASE LOCK B RECEIVE DSN COMMANDS & RELAY TO C&S C TRANSMIT ENGINEERING AND SCIENCE DATA TO DSN D REESTABLISH PHASE LOCK IF EARTH OCCULTATION OCCURS	A RECEIVE MANEUVER PARAMETERS FROM DSN & RELAY TO C&S B RECEIVE INITIATE MANEUVER COMMAND & RELAY TO C&S C RELAY MANEUVER VERIFICATION TO DSN D TRANSMIT ENGR. & SCIENCE DATA TO DSN (MODE 1)
TELEMETRY	A FORMAT S/C ENGR & SCIENCE DATA (MODE 1) B RECEIVE & FORMAT FLIGHT CAPSULE STATUS DATA	A FORMAT DATA FOR TRANSMISSION (MODE 2) •ENGINEERING & SCIENCE DATA •FLIGHT CAPSULE STATUS INFORMATION	A FORMAT S/C ENGR. & SCIENCE DATA (MODE 2, 3 OR 4) B ACCEPT & FORMAT CAPSULE STATUS DATA	A FORMAT S/C ENGR & SCIENCE DATA (MODE 1) B RECEIVE & FORMAT FLIGHT CAPSULE STATUS DATA
ANTENNA	A SWITCH TO LOW-GAIN ANTENNA B SWITCH TO HIGH-GAIN ANTENNA	A SWITCH TO MEDIUM-GAIN ANTENNA B POSITION HIGH-GAIN ANTENNA FOR INSERTION ATTITUDE C SWITCH TO HIGH-GAIN ANTENNA D REPOSITION HIGH-GAIN ANTENNA FOR ORBIT ATTITUDE	A UPDATE HIGH-GAIN ANTENNA POSITION (AS REQ'D)	A SWITCH TO MEDIUM-GAIN ANTENNA B POSITION HIGH-GAIN ANTENNA TO THRUST ATTITUDE C SWITCH TO HIGH-GAIN ANTENNA D REPOSITION HIGH-GAIN ANTENNA TO ORBIT ATTITUDE
DATA STORAGE	A RECORD MANEUVER DATA B PLAYBACK RECORDED DATA	A RECORD DATA DURING MANEUVER B PLAY BACK RECORDED DATA	A RECORD ENGINEERING & SCIENCE DATA B PLAYBACK RECORDED DATA	A RECORD MANEUVER DATA B RELAY MANEUVER DATA TO TELEMETRY
STRUCTURE & MECHANISMS	A PROVIDE STRUCTURAL SUPPORT TO S/C SYSTEMS AND CAPSULE	A PROVIDE STRUCTURAL SUPPORT TO S/C SYSTEMS AND CAPSULE	A PROVIDE STRUCTURAL SUPPORT TO S/C SYSTEMS AND CAPSULE	A PROVIDE STRUCTURAL SUPPORT TO S/C SYSTEMS AND CAPSULE
PYROTECHNICS	A FIRE SQUIBS, OPEN He VALVES B FIRE SQUIBS, OPEN START TANK PROPELLANT VALVES C FIRE SQUIBS, OPEN MAIN TANK PROPELLANT VALVES D FIRE SQUIBS, CLOSE START TANK PROPELLANT VALVES E FIRE SQUIBS, CLOSE MAIN TANK PROPELLANT VALVES F FIRE SQUIBS, CLOSE He VALVES DELETED FOR SECOND & THIRD MANEUVERS DELETED FOR THIRD MANEUVER	A FIRE SQUIBS, OPEN He VALVE	———	———
TEMPERATURE CONTROL	A PROVIDE HEAT SINK COOLING B PROVIDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING C MONITOR FLIGHT S/C TEMPERATURE	A PROVIDE HEAT SINK COOLING B PROVIDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING C MONITOR FLIGHT S/C TEMPERATURE	A PROVIDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING B MONITOR FLIGHT S/C TEMPERATURES	A PROVIDE HEAT-SINK COOLING B PROVIDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING C MONITOR FLIGHT S/C TEMPERATURE
PROPULSION	A MOVE THRUST ACTUATOR TO LOW THRUST POSITION B OPEN ENGINE VALVES, PROVIDE THRUST C CLOSE ENGINE VALVES D OPEN START TANK PROPELLANT VALVES (THIRD MIDCOURSE ONLY) E CLOSE START TANK PROPELLANT VALVES (THIRD MIDCOURSE ONLY)	A OPEN START TANK PROPELLANT VALVES B OPEN ENGINE VALVES, PROVIDE THRUST C OPEN MAIN TANK PROPELLANT VALVES D CLOSE START TANK PROPELLANT VALVES E MOVE THRUST ACTUATOR TO HIGH THRUST POSITION F CLOSE ENGINE VALVES G CLOSE MAIN TANK PROPELLANT VALVES	———	A OPEN START TANK PROPELLANT VALVES B MOVE THRUST ACTUATOR TO LOW THRUST POSITION C OPEN ENGINE VALVES, PROVIDE THRUST D OPEN MAIN TANK PROPELLANT VALVES E CLOSE ENGINE VALVES F CLOSE START TANK PROPELLANT VALVES G CLOSE MAIN TANK PROPELLANT VALVES
SCIENCE	A GATHER CRUISE DATA	———	A GATHER, ENCODE AND RELAY CRUISE/ORBITAL AND PLANETARY SCIENCE DATA. ENGINEERING DATA & MANEUVER DURING EARLY ORBITAL OPERATIONS MAY REQUIRE SCIENCE TO BE SWITCHED "OFF" FOR SHORT PERIODS	A GATHER & ENCODE CRUISE/ORBITAL SCIENCE DATA DURING MANEUVER; PLANETARY SCIENCE "OFF"
LAUNCH VEHICLE				FLIGHT CAPSULE

	0.9 FLIGHT CAPSULE CANISTER SEPARATION	0.10 FLIGHT SPACECRAFT (S/C) FLIGHT CAPSULE SEPARATION	0.15 FLIGHT SPACECRAFT ORBITAL OPERATIONS	0.16 FLIGHT SPACECRAFT MARS ORBITAL TRIM
TIME: FWD PLANETARY VEHICLE				399 DAYS
TIME: AFT PLANETARY VEHICLE				407 DAYS
LOS MOS DSN	DSN A RECEIVE ENGINEERING & SCIENCE DATA & RELAY TO SFOF B PROVIDE PV RANGE DATA C COMMAND DATA PLAYBACK D RECEIVE CANISTER SEPARATION VERIFICATION & RELAY TO SFOF E TRANSMIT BACKUP COMMAND (AS REQUIRED) SFOF A RECEIVE ENGINEERING & SCIENCE DATA & DISPLAY B RECEIVE & ANALYZE TRACK & RANGE DATA C COMPUTE TRAJECTORY PARAMETERS D RECEIVE VERIFICATION OF CANISTER SEPARATION	DSN A TRACK PV B RECEIVE ENGINEERING & RELAY TO SFOF C PROVIDE RANGE DATA D RECEIVE SEPARATION DATA VERIFICATION E RELAY BACKUP COMMANDS FROM SFOF AS REQUIRED SFOF A RECEIVE SCIENCE/ENGINEERING DATA & DISPLAY B RECEIVE & ANALYZE TRACK & RANGE DATA C RECEIVE CAPSULE SEPARATION VERIFICATION D GENERATE BACKUP COMMANDS & RELAY TO DSN (INCLUDING CAPSULE SEPARATION COMMAND)	DSN A TRACK S/C B RECEIVE ENGINEERING & SCIENCE DATA & RELAY TO SFOF C RECEIVE CAPSULE DATA D PROVIDE RANGE & TRACK DATA E COMMAND DATA PLAYBACK SFOF A RECEIVE ENGINEERING & SCIENCE DATA & DISPLAY B RECEIVE & ANALYZE TRACK & RANGE DATA	DSN A TRANSMIT MANEUVER PARAMETERS TO S/C •ROLL TURN MAGNITUDE & DIRECTION •YAW (OR PITCH) TURN MAGNITUDE & DIRECTION •MOTOR BURNTIME & ΔV B TRANSMIT INITIATE MANEUVER SEQUENCE COMMAND C RECEIVE VERIFICATION OF CORRECTION MANEUVER D TRACK S/C E PROVIDE POINTING ANGLE & RANGE DATA TO ALL DSS SFOF A COMPUTE MANEUVER PARAMETERS & TRANSMIT TO DSN B VERIFY CORRECTION MANEUVER C RECEIVE & DISPLAY ENGINEERING & SCIENCE DATA
POWER	A SWITCH TO BATTERY (NO POWER TO CAPSULE DURING OFF-SUN PERIODS) B SWITCH TO SOLAR POWER MODE AFTER MANEUVER C START CHARGING BATTERIES	A PROVIDE SOLAR POWER TO S/C & FLIGHT CAPSULE (UP TO SEPARATION)	A PROVIDE SOLAR POWER TO S/C B SWITCH TO BATTERY POWER DURING OCCULTATION (NOT FOR INITIAL 30 DAYS) C SWITCH TO SOLAR POWER MODE WHEN AGAIN ON-SUN & TURN ON BATTERY CHARGER D TRANSFER TO TRICKLE CHARGE WHEN PEAK SCIENCE LOAD IS TURNED ON. RETURN TO FULL CHARGE WHEN PEAK SCIENCE IS TURNED OFF. E TRANSFER TO BATTERY TRICKLE CHARGE FOLLOWING FULL RECHARGE	A SWITCH TO BATTERY (NO POWER TO CAPSULE DURING OFF-SUN PERIODS) B SWITCH TO SOLAR POWER MODE AFTER MANEUVER C START CHARGING BATTERIES
COMPUTING & SEQUENCE (C&S)	A RECEIVE COMMAND DECODER BACKUP OF C&S (AS REQ'D) B COMMAND MANEUVER RECORDER "ON" C COMMAND GYROS "ON" D COMMAND RADIO SWITCH TO MEDIUM-GAIN ANTENNA E COMMAND ATTITUDE CHANGE (ROLL PITCH OR YAW) F COMMAND REACQUIRE CELESTIAL REFERENCES G COMMAND SWITCH TO HIGH-GAIN ANTENNA H COMMAND MANEUVER RECORDER "OFF"; CRUISE SCIENCE MODE "ON" I COMMAND DATA STORAGE PLAYBACK J COMMAND GYROS "OFF" (AS REQ'D)	A RECEIVE COMMAND DECODER BACKUP OF C&S (AS REQ'D) B COMMAND RADIO TURN CAPSULE SUPPORT RELAY RECEIVERS "ON" C COMMAND CAPSULE DATA STORAGE RECORDER "ON" D TRANSMIT COMMAND SEPARATION SIGNAL TO CAPSULE C&S SUBSYSTEM E TRANSMIT UPDATED MANEUVERS PARAMETERS TO CAPSULE C&S (AS REQ'D) F COMMAND CAPSULE POWER FROM S/C "OFF"	A COMMAND CAPSULE SUPPORT RECORDER PLAYBACK (APPROXIMATELY 2 DAYS) B RECEIVE LIMB & TERMINATOR CROSSINGS, AND COMMAND SCIENCE SUBSYSTEM "ON" & "OFF" AS DESIRED C COMMAND BATTERY CHARGER TO TRICKLE CHARGE DURING PEAK SCIENCE LOAD. COMMAND RETURN TO FULL CHARGE WHEN PEAK SCIENCE LOAD IS TURNED OFF D COMMAND CANOPUS CONE ANGLE UPDATE E COMMAND HIGH-GAIN ANTENNA POSITION UPDATE F COMMAND DECODER BACKUP OF C&S (AS REQ'D)	A RECEIVE MANEUVER PARAMETERS FROM COMMAND DECODER BACKUP (AS REQUIRED) B COMMAND MANEUVER RECORDER "ON" C COMMAND GYROS & ACCELEROMETER "ON" D COMMAND RADIO SWITCH TO MEDIUM-GAIN ANTENNA E COMMAND S/C ATTITUDE CHANGE F COMMAND ENABLE TVC ACTUATOR & ELECTRONICS G COMMAND INSERTION MANEUVER FIRING, COUNT ΔV, AND TERMINATE H COMMAND TVC "OFF" I COMMAND ACCELEROMETER "OFF" J COMMAND REVERSE MANEUVERS K COMMAND REACQUISITION OF CELESTIAL REFERENCES (ROLL, PITCH OR YAW) L RECEIVE OPERATIONAL EVENT DATA FROM G&C M COMMAND SWITCH TO HIGH-GAIN ANTENNA N COMMAND MANEUVER RECORDER "OFF"; SCIENCE MODE "ON" O COMMAND DATA STORAGE PLAYBACK P COMMAND ACCELEROMETER & GYROS "OFF"
GUIDANCE AND CONTROL	A GYROS "ON" (AS REQUIRED) B POSITION S/C FOR SEPARATION MANEUVER (ROLL, PITCH OR GYRO) C GO TO INERTIAL "HOLD" MODE D ROLL, PITCH OR YAW BACK TO PREVIOUS ATTITUDE E REACQUIRE CELESTIAL REFERENCES; SWITCH GYRO RATE "OUT" F GYROS "OFF" (AS REQUIRED)	A MAINTAIN CELESTIAL REFERENCES DURING CAPSULE SEPARATION B GYROS ON (AS REQ'D)	A SENSE SUN OCCULTATION; SWITCH TO INERTIAL "HOLD" PITCH & YAW AXES B SENSE SOLAR PRESENCE; SWITCH TO CELESTIAL "HOLD" PITCH & YAW AXES C SENSE CANOPUS OCCULTATION; SWITCH TO INERTIAL "HOLD" MODE, ROLL AXIS D SENSE CANOPUS PRESENCE, REACQUIRE CANOPUS, SWITCH TO CELESTIAL "HOLD"; ROLL AXIS E SENSE LIMB & TERMINATOR CROSSINGS, RELAY TO C&S F UPDATE CANOPUS CONE ANGLE	A GYROS AND ACCELEROMETER "ON" B POSITION S/C FOR CORRECTION MANEUVER (ROLL, PITCH OR YAW) C INERTIAL HOLD MODE FOR ΔV (CONTROL PITCH OR YAW ATTITUDE WITH TVC, YAW BY REACTION JETS) D ENABLE TVC ACTUATOR & ELECTRONICS E YAW OR PITCH BACK TO PREVIOUS ATTITUDE F REACQUIRE SUN, SWITCH GYRO RATE "OUT" ROLL AXIS G ROLL BACK TO PREVIOUS ATTITUDE H REACQUIRE CANOPUS, SWITCH GYRO RATE "OUT" ROLL AXIS I ACCELEROMETER MEASURES ΔV INCREMENT J TVC AND ACCELEROMETER "OFF"
RADIO	A TRANSMIT ENGINEERING & SCIENCE DATA TO DSN B REESTABLISH PHASE LOCK (IF REQUIRED) C TRANSMIT PLAYBACK DATA ON COMMAND FROM DSN D TRANSMIT CANISTER SEPARATION VERIFICATION TO DSN E RECEIVE COMMANDS FROM DSN (AS REQUIRED)	A RECEIVE CAPSULE MANEUVER PARAMETERS FROM DSN & RELAY TO C&S B RECEIVE SEPARATION COMMAND & RELAY C SWITCH RELAY RECEIVER "ON" D RECEIVE & RELAY CAPSULE DATA	A RECEIVE CAPSULE DATA & RELAY TO CAPSULE DATA RECORDER B TURN RELAY RECEIVER "OFF" C TRANSMIT CAPSULE DATA PLAYBACK TO DSN D EARTH OCCULTATION - REESTABLISH PHASE LOCK E TRANSMIT ENGINEERING & SCIENCE DATA TO DSN	A RECEIVE MANEUVER PARAMETERS FROM DSN & RELAY TO C&S B RECEIVE INITIATE MANEUVER COMMAND & RELAY TO C&S C RELAY MANEUVER VERIFICATION TO DSN D TRANSMIT ENGR. & SCIENCE DATA TO DSN (MODE 1)
TELEMETRY	A FORMAT S/C ENGR. & SCIENCE DATA (MODE 2, 3 or 4) B ACCEPT & FORMAT CAPSULE STATUS DATA	A ACCEPT & FORMAT CAPSULE DATA B FORMAT S/C ENGINEERING & SCIENCE DATA	A ACCEPT & FORMAT CAPSULE DATA (APPROX. FOR 2 DAYS) B FORMAT S/C ENGINEERING & SCIENCE DATA (MODES 2, 3 OR 4)	A ACCEPT AND FORMAT DATA FOR TRANSMISSION (MODES 2, 3 OR 4)
ANTENNA	A SWITCH TO MEDIUM-GAIN ANTENNA B SWITCH TO HIGH-GAIN ANTENNA C REPOSITION HIGH-GAIN ANTENNA (IF REQUIRED)	A RECEIVE CAPSULE DATA OVER RELAY ANTENNAS	A UPDATE HIGH-GAIN ANTENNA POSITION (AS REQ'D) B RECEIVE CAPSULE DATA OVER RELAY DIPOLE ANTENNA	A SWITCH TO MEDIUM-GAIN ANTENNA B POSITION HIGH-GAIN ANTENNA TO THRUST ATTITUDE C SWITCH TO HIGH-GAIN ANTENNA D REPOSITION HIGH-GAIN ANTENNA TO ORBIT ATTITUDE
DATA STORAGE	A RECORD MANEUVER DATA B PLAYBACK MANEUVER DATA TO TELEMETRY	A RECORD CAPSULE DATA B RECORD ENGINEERING AND SCIENCE DATA	A RECORD CAPSULE DATA/OR PLAYBACK OF CAPSULE MISSION B RECORD S/C ENGINEERING AND SCIENCE DATA	A RECORD MANEUVER DATA B PLAY BACK RECORDED DATA TO TELEMETRY
STRUCTURE & MECHANISMS	A PROVIDE STRUCTURAL SUPPORT TO S/C SYSTEMS AND CAPSULE	A PROVIDE STRUCTURAL SUPPORT TO S/C SYSTEMS	A PROVIDE STRUCTURAL SUPPORT TO S/C SYSTEMS	A PROVIDE STRUCTURAL SUPPORT TO S/C SYSTEMS
PYROTECHNICS	A FIRE EED, SEPARATE CAPSULE CANISTER	———	———	———
TEMPERATURE CONTROL	A PROVIDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING B MONITOR FLIGHT S/C TEMPERATURES	A PROVIDE HEAT-SINK COOLING B PROVIDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING C MONITOR FLIGHT S/C TEMPERATURE	A PROVIDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING B MONITOR FLIGHT S/C TEMPERATURES	A PROVIDE HEAT-SINK COOLING B PROVIDE PASSIVE & SEMIPASSIVE RADIATIVE COOLING C MONITOR FLIGHT S/C TEMPERATURE
PROPULSION	———	———	———	A OPEN START TANK PROPELLANT VALVES B MOVE THRUST ACTUATOR TO LOW THRUST POSITION C OPEN ENGINE VALVES, PROVIDE THRUST D OPEN MAIN TANK PROPELLANT VALVES E CLOSE ENGINE VALVES F CLOSE START TANK PROPELLANT VALVES G CLOSE MAIN TANK PROPELLANT VALVES
SCIENCE	A GATHER, ENCODE & RELAY CRUISE/ORBITAL SCIENCE DATA DURING MANEUVER	A GATHER, ENCODE & RELAY CRUISE/ORBITAL SCIENCE DATA B PLANETARY SCIENCE DATA "OFF"	A GATHER, ENCODE & RELAY PLANETARY SCIENCE DATA B GATHER, ENCODE & RELAY CRUISE/ORBITAL SCIENCE DATA C TURN SCIENCE "ON" & "OFF" PER C&S COMMANDS	A GATHER & ENCODE CRUISE/ORBITAL SCIENCE DATA DURING MANEUVER; PLANETARY SCIENCE "OFF"
LAUNCH VEHICLE		A TURN "ON" INTERNAL POWER B ESTABLISH RADIO LINK WITH S/C C UPDATE CAPSULE PROGRAM (AS REQUIRED) D RECEIVE SEPARATION SIGNAL FROM S/C & FIRE EED'S E PERFORM ATTITUDE ALIGNMENT MANEUVER AS REQ'D F PROVIDE DEBOOST ΔV G TRANSMIT ENGINEERING & SCIENCE DATA TO S/C	A TRANSMIT ENGINEERING & SCIENCE DATA TO EARTH FROM MARTIAN SURFACE (1600 BPS) B TRANSMIT ENGINEERING & SCIENCE DATA TO ORBITING S/C	

Figure 1.1.6-1: NOMINAL FLIGHT SEQUENCE


165-INCH DIA HIGH GAIN ANTENNA, 2 AXIS DRIVE ON 84° & 174° CLOCK ANGLES (DEPLOYED POSITION)
HINGE MOTOR
SWIVEL MOTOR
SHROUD EXTERIOR
DYNAMIC ENVELOPE
STOWED POSITION
SOLAR ARRAY
-X AXIS
+X AXIS
SPACECRAFT ℄
SECTION A-A
+Z AXIS
A
STAR TRUSS MEMBER (TYP. 16 PLACES)
SOLAR ARRAY EDGE & DYNAMIC ENVELOPE
TELECOMMUNICATIONS SUBSYSTEM BAY
PROPELLANT PRESSURANT TANK, 9.0 CUBIC FEET HELIUM, 3500 PSIA NOMINAL, 31.0 DIA SPHERE (TYP 2 PLACES, DIAMETRICALLY OPPOSED)
TELECOMMUNICATIONS SUBSYSTEM BAYS
LOW GAIN ANTENNA, 2 PL DIAMETRICALLY OPPOSED (DEPLOYED POSITION) CLOCK ANGLES = 122° & 302° CONE ANGLES = 95° & 95°
RELAY DIPOLE ANTENNA (DEPLOYED POSITION) ELEMENT ORIENTATION: CLOCK ANGLE = 141° CONE ANGLE = 51° AIM POINT: CLOCK ANGLE = 39° CONE ANGLE = 67°
+Y AXIS
+PITCH
+YAW
PROPULSION LOUVER ASSY (TYP 4 PL)
FLIGHT SPACECRAFT/FLIGHT CAPSULE FIELD JOINT, 160-INCH DIA. BOLT CIRCLE
MAIN PROPELLANT TANK, 47.7 CUBIC FEET MONOMETHYL HYDRAZINE, 245 PSIA NOMINAL, 54.0 DIA. SPHERE (TYP 2 PLACES, DIAMETRICALLY OPPOSED)
COMPUTING & SEQUENCING SUBSYSTEM BAY
36-INCH DIA. MEDIUM GAIN ANTENNA, 1 AXIS DRIVE ON 174° CLOCK ANGLE (DEPLOYED POSITION)
ENGINE START TANK, 2.2 CUBIC FEET MONOMETHYL HYDRAZINE, 245 PSIA NOMINAL, 20.0 DIA x 32.0 CYLINDER
COMPUTING & SEQUENCING SUBSYSTEM BAY
MAIN PROPELLANT TANK, 47.7 CUBIC FEET NITROGEN TETROXIDE, 245 PSIA NOMINAL, 54.0 DIA SPHERE (TYP 2 PLACES, DIAMETRICALLY OPPOSED)
INCREASING CLOCK ANGLE
+Z AXIS
TO CANOPUS
PLAN VIEW
SPACE CONFIGURATION
(LOOKING TOWARDS THE SUN)
165-INCH DIA HIGH GAIN ANTENNA, 2 AXIS DRIVE ON 84° & 174° CLOCK ANGLES (DEPLOYED POSITION) 2
SPACECRAFT/SHROUD SEPARATION JOINT (TYP 8 PLACES)
RELAY HELIX ANTENNA (STOWED POSITION)
RELAY HELIX ANTENNA (DEPLOYED POSITION) CLOCK ANGLE = 195° CONE ANGLE = 126° 2
TELECOMMUNICATIONS SUBSYSTEM BAY
ENGINE START TANK, 2.2 CUBIC FEET NITROGEN TETROXIDE, 245 PSIA NOMINAL, 20.0 DIA x 32.0 CYLINDER
COARSE SUN SENSOR (TYP 8 PL. DARK & LIGHT SIDE OF SOLAR ARRAY, 45° OFF AXES)
SCIENCE SUBSYSTEM BAY
ULTRA-VIOLET SPECTROMETER SCAN PLATFORM 2-AXIS DRIVE CLOCK ANGLE = 247.5° CONE ANGLE = 90°
SCIENCE SCAN PLATFORM (STOWED POSITION)
SCIENCE SCAN PLATFORM (DEPLOYED POSITION)
-Y AXIS
SCIENCE SCAN PLATFORM, 3-AXIS DRIVE (SCANNING POSITION) 3π STERADIAN COVERAGE CENTERED ON CLOCK ANGLE 270°, CONE ANGLE 90° 2
PLANETARY VEHICLE CONNECTOR (TYP 2 PL) (REF)
ION CHAMBER
PLASMA PROBE (2 PLACES DIAMETRICALLY OPPOSED) CLOCK ANGLES = 137.5° & 317.5° CONE ANGLES = 0° 0°
AIM POINTS: 141° & 321° 10° 10°
POWER SUBSYSTEM BAYS (TYP. 4 PL.)
SHROUD EXTERIOR (REF)
COSMIC DUST DETECTOR CLOCK ANGLES = 90° & 270° CONE ANGLE = 90°
GUIDANCE & CONTROL ELECTRONICS ASSY
SECTION B-B
DEPLOYMENT PIVOT
SWIVEL MOTO
AZIMUTH MOTO
HINGE MOT
SCIENCE (SCANNING
SOLAR ARRAY
SECTION C-C
HINGE
MEDIUM
DEPLOY
SECTION D-D

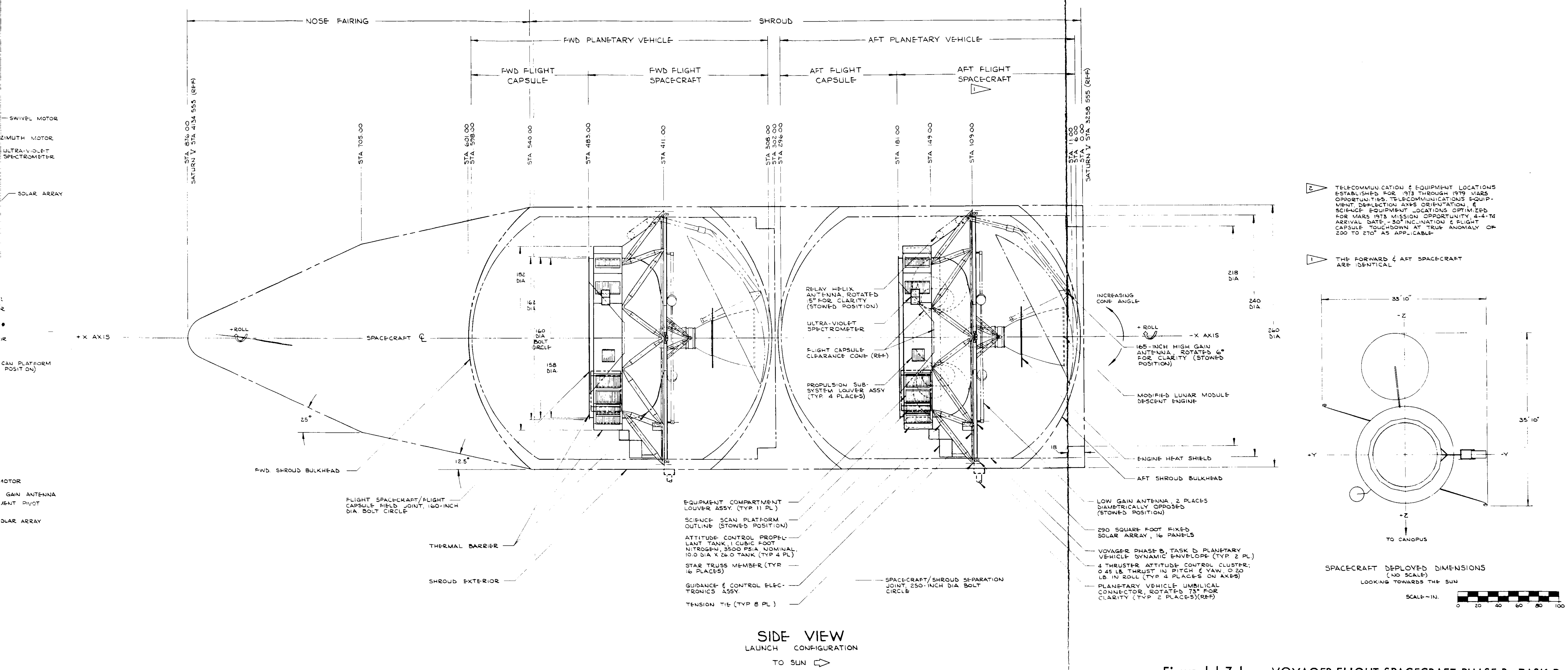


Figure 1.1.7-1: VOYAGER FLIGHT SPACECRAFT PHASE B, TASK D

FOLDOUT FRAME 3

FOLDOUT FRAME 4

FOLDOUT FRAME 5

4) Unobstructed views for all sensors and antennas where required

5) Growth for later mission with minimum modifications.

1.1.7.1 Design Configuration Parameters

In developing the design a number of parameters identifying subsystem requirements, space limitations, and other requirements were derived. It was necessary to satisfy these requirements in developing the configuration concept. The major configuration parameters follow:

Science Payload	371 pounds
Overall Diameter	240 inches
Overall length	170 inches
Capsule Attachment Diameter	160 inches
Solar Array Area	290 square feet
High-Gain Antenna Diameter	13.7 feet
Equipment Packaging Area	36 square feet
Engine Type	Lunar Module Descent Engine
Usable Propellant Capacity	12,873 pounds **(See Table 2.2.1-1, page 2-4)**
Helium Pressurant Volume	19 cubic feet
Structural Arrangement	Truss

1.1.7.2 Mars 1973 Baseline Design

The baseline vehicle is 170 inches long and 240 inches in diameter. The concept employs an external support truss and an internal propulsion truss. Capsule support is provided by the upper portion of the propulsion system truss. Boost loads are carried at eight points at the booster shroud through the external truss and into the propulsion system truss. No meteoroid shielding is specifically provided.

The external support truss is used to support the attitude control system, a guidance and control bay, and deployable equipment such as the science scan platform and low, medium, and high gain antennas. The attitude control nozzles are mounted on the outer circumference on the Y and Z axes. A fixed bay containing all guidance and control equipment is mounted on the outer circumference providing an unobstructed field of view for the Canopus sensor. All other electrical and electronic equipment is mounted on plates secured between the capsule supporting ring and the upper ring of the support truss. Cable trays form the upper and lower closures. The ring is divided into 16 bays, six of which contain the basic electronic equipment and two of which are provided for science and growth. The remaining bays can be easily modified to accept equipment. Typical design of the electronic modules is shown in Figure 1.1.7-2. The circular solar panel spans the distance between the outer extremities of the external support truss and the inner portion of the propulsion truss.

The propulsion system truss supports four main propellant tanks (54 inches in diameter), two start tanks, and two propellant pressurant tanks. The LMD engine is supported by dual truss supports mounted from the inner portion of the propulsion truss.

The baseline utilizes modular construction to allow parallel construction and testing, and to generally improve overall spacecraft flexibility. The modular construction is highlighted by the exploded view shown in Figure 1.1.7-3.

The basic module is composed of the external support truss, the primary spacecraft support ring, and the deployable components. The equipment bays and cable tray assembly form a second module which attaches to the primary spacecraft support ring. The removable propulsion module forms the third module and contains and supports the capsule support structure, the main propellant tanks, the propellant pressurant tanks, the start tanks, the engine, and all associated plumbing and wiring. This module fits inside the primary spacecraft support ring and joins that assembly at the eight apexes of the support truss. The equipment support assembly is stabilized by joining the upper cable tray to the upper portion of the propulsion module. The guidance and control system is installed and aligned with the spacecraft. The attitude control nozzles are mounted on the external support truss on the Y and Z axes and the four nitrogen tanks are installed in pairs on the Z axis in the equipment module. The solar panel module connects between the outer circumference of the external support truss and the inside of the propulsion module.

A detailed weight statement is shown in Table 1.1.7-1 for Case A (5000 pounds) and Case B (6000 pounds) capsule weights. Case B was used for the baseline configuration definition. If a Case A capsule weight is used, a margin of 1930 pounds is available per planetary vehicle.

1.1.7.3 Configuration Development

A number of configuration trade studies was undertaken in the development of the baseline design. In addition, sensor location studies and alternate engine investigations were made. These are summarized as follows:

Length to Diameter Trade -- The relationship of the spacecraft length to diameter was of interest because of the dynamic envelope limitations within the shroud diameter as a function of the available spacecraft length which would allow two spacecraft to be installed and still meet the limitation on available shroud length. A parametric study was made of spacecraft which varied in length and diameter but which still met the requirements for required volume to house the Voyager equipment. This study is summarized in Figure 1.1.7-4. In the diameter range of greatest interest (140 to 180 inches) total spacecraft gross weight changes very little and

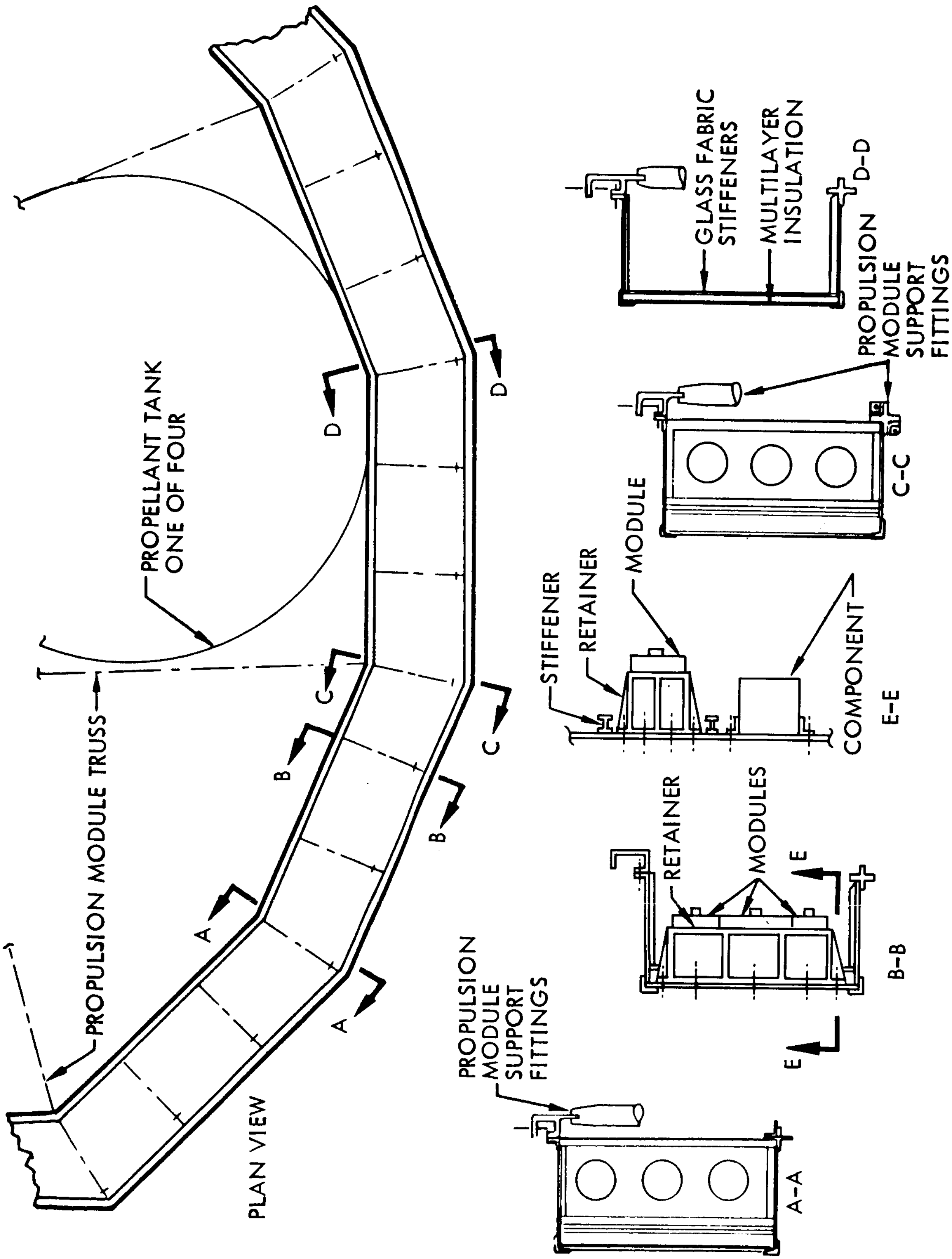


Figure 1.1.7-2: ELECTRONIC MODULES

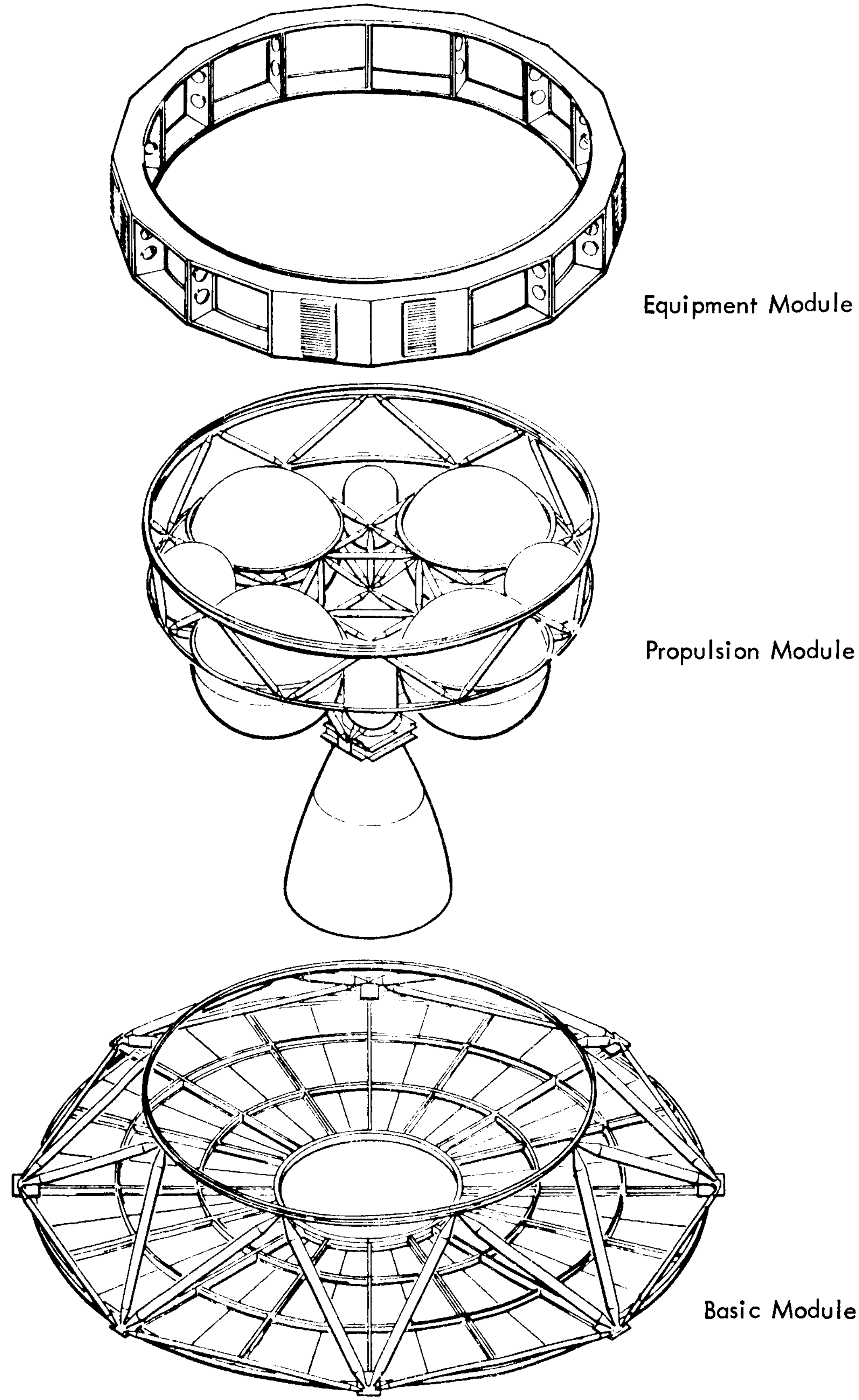


Figure 1.1.7-3: BASELINE MODULES

	CASE B			CASE A		
			WEIGHTS (Pounds)			WEIGHTS (Pounds)
STRUCTURE			539			
Capsule Support Truss		129				
Upper Ring/Field Joint Flange	69					
Compression Struts & Fittings (16)	54					
Miscellaneous	6					
Star Truss		318				
Upper Ring	130					
Compression Struts & Fittings	158					
Tension, Ties and Fittings	15					
Miscellaneous	15					
Special Purpose Fittings		92				
Capsule Truss/Star Truss	50					
Tiedown/Separation Pads	20					
Miscellaneous	22					
PROPULSION			1,799			
Lunar Module Descent Engine		409				
Fuel System		438.5				
Tanks (Main)	271.5					
Tanks (Bellows)	38.5					
Tank Support	45.0					
Plumbing	77.5					
Miscellaneous	6.0					
Oxidizer System		464.5				
Tanks (Main)	271.5					
Tanks (Bellows)	38.5					
Tank Support	71.0					
Plumbing	77.5					
Miscellaneous	6.0					
Pressurization System		475				
Tanks	440					
Tank Supports	4					
Plumbing	31					
Engine Supports		12				
EQUIPMENT & INSTRUMENTATION			2595			
Structure (Equipment Component and Thermal)		189				
Radiator Plates	77.8					
Stiffeners	14.9					
Shear Webs & Chords	18.9					
Cable Tray	52.7					
Miscellaneous	24.7					
Guidance & Control		208				
Sensors	20					
Computing & Sequencing	69					
Inertial Reference Unit	28					
Attitude Control Electronics	12					
Accelerometer Unit	5					
Gimbal Actuators – LMDE (2)	36					
Guid. & Control Electronics Ass'y. & Support	38					
Instrumentation		476				
Radio	100					
Telemetry	34					
Data Source	134					
Antennas	208					
Electrical Power		573				
Solar Array	250					
Batteries	249					
Power Control	74					
Electric Networks		324				
Networks	265					
Pyrotechnics	59					
Temperature Control		184				
Insulation	107.6					
Louvers & Mechanism	19.5					
Paint	23.3					
Ground Cooling	12.0					
Miscellaneous	21.6					
Attitude Control		124				
Tanks	93.0					
Plumbing	18.4					
Thrusters	9.6					
Supports	3.0					
Science Equipment		146				
Scan Platforms	111					
Scan Platform Supports	32					
Cruise Instrument Support	3					
Science Payload		371				
Science Instruments	301					
Data Automation Electronics & Misc.	70					
CONTINGENCY			257			
TOTAL DRY SPACECRAFT			5,190			
RESIDUALS			379			
Propellants		338				
Tanks	193					
Lines & Engines	52					
Start, Stop & Leakage	80					
Loading Allowance	13					
Pressurants		40				
Attitude Control System Fluids		1				
TOTAL INERT SPACECRAFT			5,569			5,569
USABLE PROPELLANT			10,951			10,021
Fuel		4,190		3,830		
Oxidizer		6,700		6,130		
Nitrogen		61		61		
TOTAL SPACECRAFT AT LIFTOFF			16,520			15,590
CAPSULE			6,000			5,000
TOTAL PLANETARY VEHICLE AT LIFTOFF			22,520			20,590
ADAPTER PROVISIONS			240			240
TOTAL PLANETARY VEHICLE + ADAPTER			22,760			20,830

Table 1.1.7-1: DETAILED WEIGHT STATEMENT (1973 VOYAGER)

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

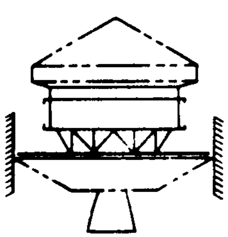
The primary structure for the bus must provide support to the electronic packages, transfer capsule loads to the adapter and perform other secondary structural functions.

Boost loads are the significant design parameter. The purpose of the study is to identify an optimum bus diameter within the restrictions of structural efficiency and the dynamic envelope.

Design ground rules are:

1) Semimonocoque design using non compression-buckling, shear-resistant skin.
2) Circumferential and longitudinal stiffening.
3) Concentrated loads at eight points with doublers to redistribute the concentrated loads.
4) All aluminum bus.
5) Combined load condition using limit load factors of 4.75 axial and 2.0 lateral.
6) An ultimate factor of safety of 1.25.
7) Constant internal bus volume (575 ft^3).
8) Capsule weight = 7000 lb

CONFIGURATION USED FOR TRADE

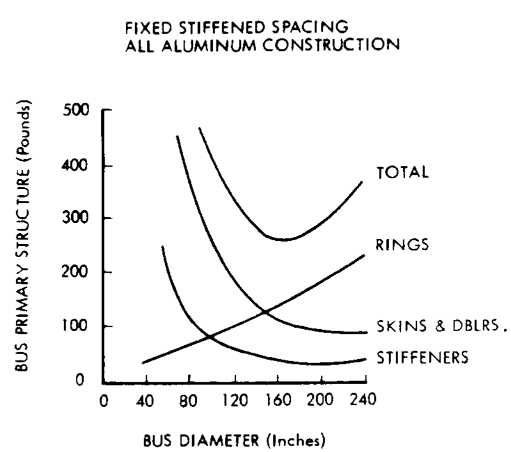


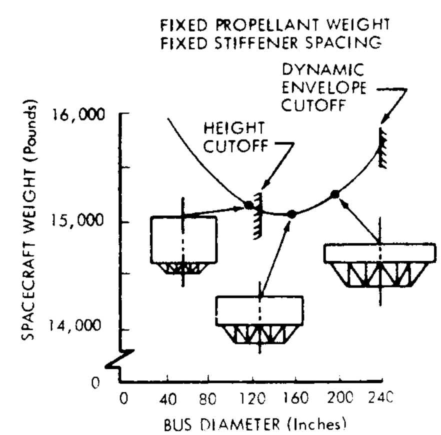
TRADE STUDY NUMBER & TITLE: BUS LENGTH /DIAMETER EFFECTS

MATRIX OF DESIGN APPROACH

	POUNDS DIAMETER = 80 IN.	POUNDS DIAMETER = 120 IN.	POUNDS DIAMETER = 160 IN.	POUNDS DIAMETER = 200 IN.
DIAMETER DEPENDENT WEIGHT				
Structural and Mechanical	1140	940	915	[illegible]
Cabling	180	220	260	300
Temperature Control	135	140	170	210
Propellant Tankage	560	520	430	430
Reaction Control	285	230	230	230
Weight Contingency	115	100	100	110
TOTAL (DIA. - DEPENDENT WEIGHT)	2415	2150	2105	2270
NONDIAMETER DEPENDENT WEIGHT	3245	3245	3245	3245
PROPELLANT	9750	9750	9750	9750
SPACECRAFT GROSS WEIGHT	15410	15145	15100	15265

This weight breakdown has been computed considering all parameters which change as a function of bus diameter.





SELECTION

NOTE:

A definite minimum weight exists in bus primary structural weight, however, spacecraft weight is relatively insensitive to diameter in the range of interest.

SELECTED APPROACH

Select and diameter between 140 and 180 inches for minimum weight.

Figure 1.1.7-4: BUS LENGTH/DIAMETER TRADE STUDY is within the bucket of the curve. Therefore, the length and diameter of the spacecraft were not selected on the basis of minimum weight, but were selected primarily on the basis of the optimum equipment installation which would best meet the other major design objectives such as sensor field of view and structural efficiency.

Separate Versus Integral Adapter -- A study was made of separate versus integral adapter to determine if a significant advantage of payload in Mars orbit was available with either system. These results are summarized on Figure 1.1.7-5.

The study indicates the total payload in Mars orbit is 67 pounds greater with the integral adapter than could be achieved with a separate adapter. While this is not a decisive margin in favor of the integral adapter, other considerations, such as structure duplication required for equipment installation with the separate adapter, also favored it.

Propellant Settling -- In a zero-gravity environment, sufficient propellant must be available at the tank outlets to start the engines and settle the propellant. Figure 1.1.7-6 shows some of the concept studies with advantages and disadvantages of the various systems. Several systems for starting liquid fueled engines in space have been used. One employs screens in the main fuel and oxidizer tanks. This system may not work after small propulsion efforts because the propellant may not be properly settled. The propulsion settling can be performed with a separate engine and tanks or with one engine with separate starting tanks. Bellows were used instead of bladders for positive fuel expulsion because of the possibility of helium leakage through bladders at zero g over long periods of time. The starting system can also contain enough fuel to do the midcourse and orbit trim maneuvers, but this takes rather large tanks that are beyond the present state of the art. Separate start tanks were selected for the baseline system.

During orbit trim powered flight, an unstable slosh mode exists. The detail requirements for baffling must be determined from an analysis of this condition.

Alternate Engine Investigation -- A study was made of the feasibility of use of the Atlas-Agena or Transtage engines in place of the lunar module descent engine (LMDE) used on the baseline design. Physically these engines are interchangeable with only minor modifications. Figure 1.1.7-7, an overlay of the Transtage and Agena engine installations, shows they are easily contained within the envelope of the baseline LMDE. Figure 1.1.7-8 is a comparison of the three engines showing gimbal points and fueling ports. Additional information on the alternate engine utilization is contained in Section 1.2.9.

Sensor Location -- A difficult configuration problem was the location of the various sensors to obtain the proper unobstructed field of view. Polar plots were made of the required fields of view for each of the sensors and the shadowgraph showing the portion of the field of view of the sensor which is obstructed by the spacecraft. These plots were superimposed to determine the locations for five of the sensors as follows:
The high gain antenna is deployed from its position under the solar panels to a position of 84 degrees cone angle at 174 degrees clock angle.

The relay link antenna is fixed mounted on the planetary vehicle bus at a clock angle of 195 degrees.

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE: SEPARATE VERSUS INTEGRAL PLANETARY VEHICLE ADAPTER	SELECTION

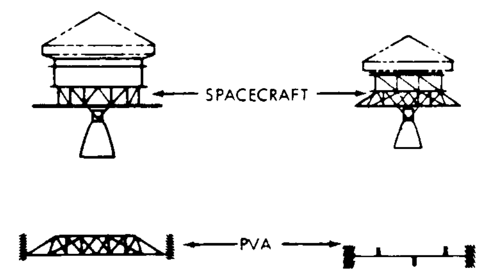
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

The planetary vehicle adapter provides a structural load path for transferring planetary vehicle loads to the Saturn V shroud during launch. This load path can be provided either by a separate structural system which bridges between the shroud and planetary vehicle or by the planetary vehicle structure, depending on configuration.

For this trade the following requirements were defined:

1. Spacecraft in Mars orbit (less structure and propulsion) = 2500 lb
2. Flight capsule = 5000 lb
3. Spacecraft propulsion capable of 1950 M/S

MATRIX OF DESIGN APPROACH



	SEPARATE P/A	INTEGRAL P/A
Spacecraft in Mars Orbit (Less structure and propellant)	2500 lb	2500 lb
Spacecraft Structure	1029	1095
Propulsion Inerts	2230	2240
Usable Propellant (1950 M/S)	9734	9804
Spacecraft	15493	15640
Flight Capsule	5000	5000
Planetary Vehicle	20493	20640
Adapter and Modification to Saturn V Shroud	523	246
Planetary Vehicle + Planetary Vehicle Adapter	21016	20886

Separate P/A

PRO

1. Provides minimum weight spacecraft structure.
2. Provides minimum planetary vehicle weight.

CON

1. Requires 1 additional major structural system.
2. Yields maximum weight at launch

Integral P/A

PRO

1. Adapter provisions lighter by 277 lb.
2. Total P.V. lighter by 130 lb.
3. Payload to Mars orbit for given booster greater by 61 lb.
4. Minimum structural system.

CON

1. Propulsion weight 80 lb. Heavier (70 lb propellant, 10 lb inerts)

SELECTION

Weight differences are small and of secondary importance in selection. Selection based on minimum number of major structural systems and integration of dual purpose structure.

SELECTED APPROACH

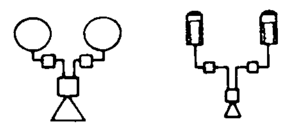
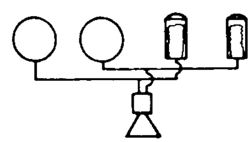
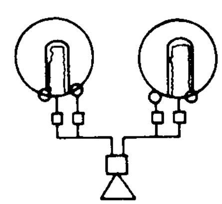
Integral Adapter

Figure 1.1.7-5: ADAPTER TRADE STUDY

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE	PROPULSION STARTING SYSTEM CONCEPT SELECTION
			MATRIX OF DESIGN APPROACH

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

1. Provide storage capability for propellants, nitrogen tetroxide and Aerozene 50 or MMH.
2. Provides liquid propellant to the engine inlet for at least six engine starts under zero-g conditions.
3. Provide liquid propellant to the engine inlet at all times while the engine is operating (approx. 0.0375 to 2 g's).
4. Provide storage capabilities as follows (pounds total propellants):
   Main tanks: 12,325 lb
   Start Tanks: 300 lb
5. Design requirements (approximate):
   Working Press: 240 psia
   Temperature: 5 - 43°C
6. Design to be compatible with sterilization environment.
7. Design to be compatible with space vacuum and thermal environment.
8. Provide system configuration that no single failure will disable.

Concept	ADVANTAGES	DISADVANTAGES
1. Free surface tanks. Separate start system. 	1. Simple concept. 2. Eliminates switching propellant feed after starting engine. 3. Small start propellant volume bellows is within state of art.	1. Requires more functional components than other concepts. 2. Heavy. 3. Free surface in main tanks will slosh, aggrevating vehicle attitude control and science sensor platform stability problems. 4. Requires at least 2 rocket engines.
2. Same as 1 above except starting system size increased to include orbit trim and midcourse maneuvers.	1. Same as concept 1 above except 3. 2. Provides least free surface area (minimizes slosh).	1. Same as concept 1 above except slosh is minimized. 2. Bellows is large and fabrication exceeds state of art.
3. Single rocket with separate start tanks. 	1. Only one rocket engine required. 2. Start tanks sized to include propellant for starts and 2nd and 3rd midcourse brings bellows size within state of art. 3. Fewer functional components required than concepts 1 and 2. 4. Lightest and simpler than concepts 1 or 2.	1. Slosh aggravates attitude control and platform stability problems.
4. Single rocket with integral start tanks. 	1. Only one rocket required. 2. Fewer functional components required. 3. Light weight.	1. Requires complex valving arrangement. 2. Complex tank design. 3. Largest tank volumes and concentrated weights.

SELECTION

Concept 3

1. Utilizes vehicle envelope efficiently.
2. Provides good weight distribution.
3. Simple plumbing.
4. Lightest.
5. Only one engine required.
6. Bellows within state of art.

SELECTED APPROACH

Concept 3, single rocket with separate start tanks.

Figure 1.1.7-6: PROPULSION STARTING TRADE STUDY

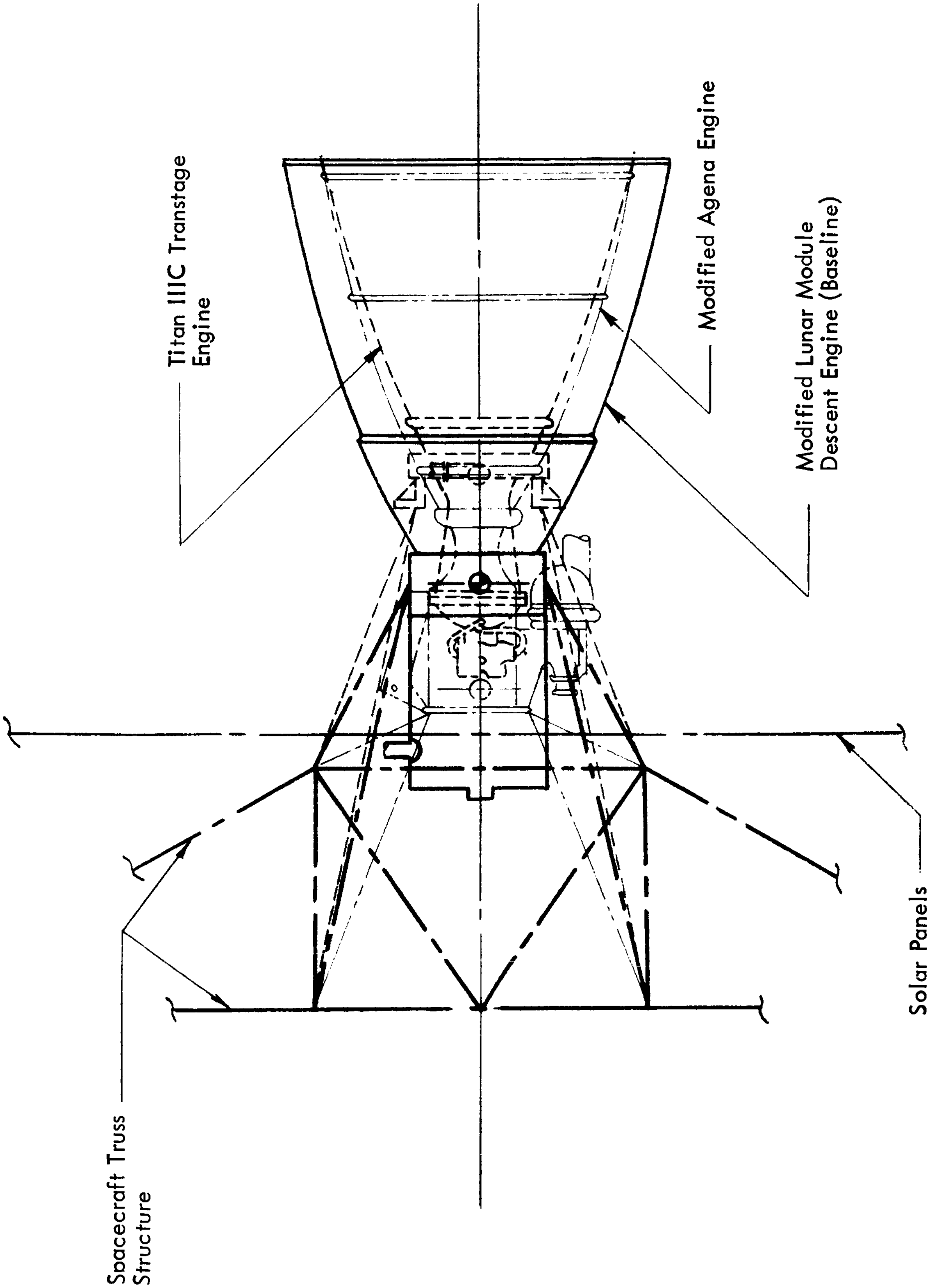


Figure 1.1.7-7: TRANSTAGE AND AGENA ENGINE INSTALLATION OVERLAY

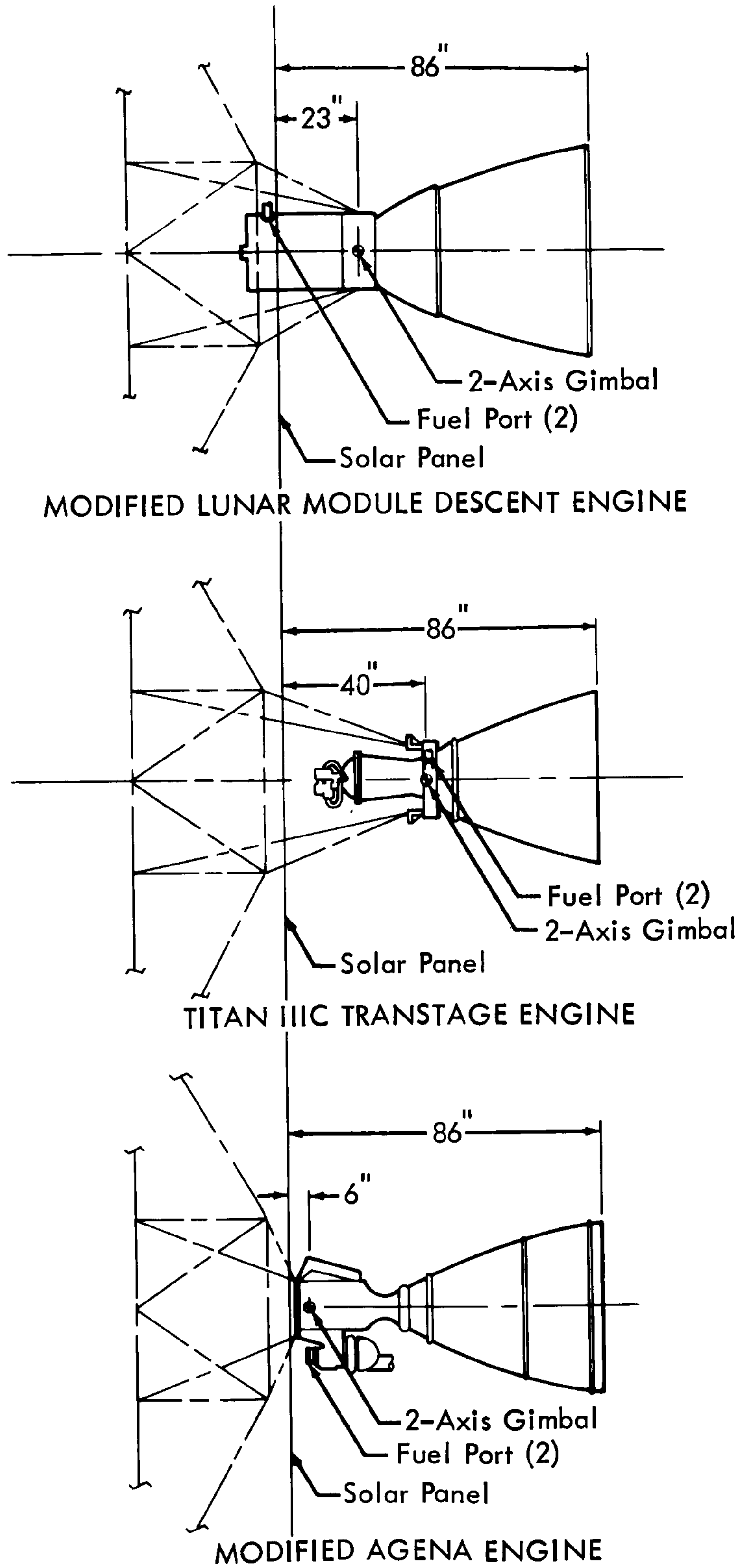


Figure 1.1.7-8: ALTERNATE ENGINE INSTALLATIONS

The science scan platform number 1 is deployed from a position between the solar panels and the capsule to a position overlooking the capsule at a clock angle of 270 degrees.

The ultraviolet spectrometer is mounted on a separate scan platform with a nominal position of 247.5 degrees clock angle and 90 degrees cone angle.

The Canopus sensors are mounted at a clock angle of 0 degree on the +Z axis at the extremity of the solar panel to minimize interference.

As an example, the shadowgraph and required fields of view developed for the high gain antenna are shown in Figure 1.1.7-9.

Equipment Module Trade Study -- A trade study involving the electronic equipment modules was made to determine the effects of various methods of equipment support on the structural concept and the thermal effects. The results of the study are shown on Figure 1.1.7-10. The study led to the selection of the baseline electronics equipment module concept shown on Figure 1.1.7-2.

1.1.7.4 Alternate Configurations

Two major alternate configurations were investigated during the study. These were a two-propellant tank design and a semimonocoque design.

Two-Tank Spacecraft Configuration -- The spacecraft configuration is to a great extent determined by the propellant tankage arrangement. To minimize the number of propellant tanks, simplify the plumbing, facilitate simultaneous filling and emptying of the tanks, and make a more compact vehicle, a two-tank configuration was designed (see Figure 1.1.7-11).

The propellant for the main propulsion system is contained in two spherical tanks of equal volume. To retain the center-of-gravity location coincident with the geometric center as the propellant is used, the tanks are located off center in proportion to their weight differences. Each tank is supported by a skirt attached at four points to a simple truss system. Start tanks and helium pressurant tanks are also located to minimize center-of-gravity movement with propellant use. Since the propulsion system is integrated with the main truss, it cannot be removed as a module. The adapter is integral; the truss system spans the entire 260-inch diameter and attaches directly to the shroud. The capsule is also supported by this truss.

The electronics are assembled as a separate module. The module is U-shaped and is installed over the outside of the truss structure from the engine end. This arrangement places all the electronic cold plates in a position to act as meteoroid protection for the propellant tanks.

Considerable growth is possible with this configuration. For example, a high resolution camera as large as 12 feet long and 40 inches in diameter can be accommodated without revision to the structure. Sixty square feet of electronic cold plate is available; this area can be doubled if required.

The science scan platform, the high gain antenna, and the guidance and control packages are all an integral part of the electronics module. The 290 ft^2 fixed

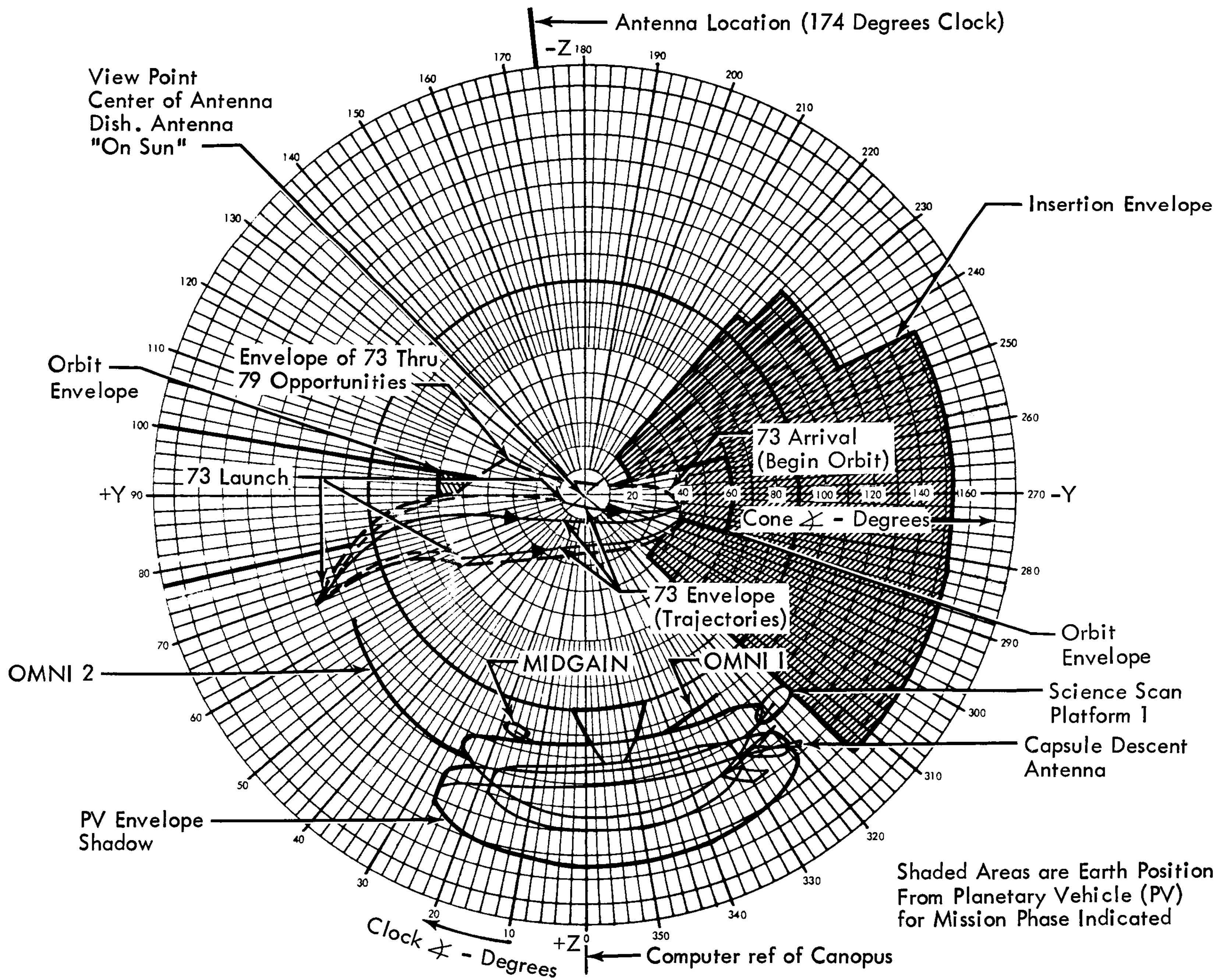


Figure 1.1.7-9: HIGH GAIN ANTENNA SHADOWGRAPH

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT

TRADE STUDY NUMBER & TITLE: SPACECRAFT EQUIPMENT SUPPORT VERSUS STRUCTURAL CONCEPT

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

The primary structure of the spacecraft bus must provide support for the electronic packages as well as perform its other functions.

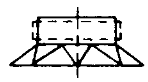
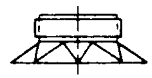
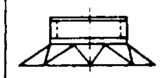
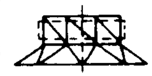
The loads encountered during the boost phase are the significant design parameter for the spacecraft bus structure.

Access to and removal of electronic packages is required for maintenance.

Ground rules for this trade study are:

1) 160-inch diameter bus
2) Load factors of:
   $N_x = + 5.94, -2.97$ ultimate
   $N_{yiz} = \pm 2.5$ ultimate
3) Flight capsule weight of 5000 lb.
4) Semimonocoque structure is shear resistant.
5) Equipment accessability is a requirement.

MATRIX OF DESIGN APPROACH

	Config. 1	Config. 2	Config. 3	Config. 4
				
Description				
Bus	Semimonocoque	Semimonocoque	Semimonocoque	Semimonocoque
Electronic package shear-carrying	NO	YES	YES	NO
Location	Outside	Outside	Inside	Outside
Weight Breakdown (Bus structure only -Excludes propulsion support structure)	523	470	427	434
ADVANTAGES	1. Modularization of electronics (Remove entire unit). 2. Structural integrity W/O electronic module. 3. Good wiring access	1. Structure use of chassis for primary struct. loads.	1. Structure use of chassis for primary struct. loads. 2. Lightest bus structural system.	1. Modularization of electronics (remove entire unit.) 2. Structural integrity W/O electronic module. 3. Good wiring access
DISADVANTAGES	1. Heaviest bus structure	1. Requires close fit holes working through frequent removals. 2. Chassis removal leaves S/C unsupported for shear loads.	1. Requires close fit holes working through frequent removals. 2. Chassis removal leaves S/C unsupported for shear loads.	1. Slightly heavier than monocoque with a structural cold plate.

SELECTION

Reliability 4,1,3,2

Maintainability 4,1,2,3

Thermal Efficiency 2,1,4,3

Accessibility 4,1,2,3

Structural Integ. 1,4,3,2

Weight 3,4,2,1

SELECTED APPROACH

Configuration 4

Based on relaibility, maintainability, accessibility, and structural integrity.

Figure 1.1.7-10: EQUIPMENT MODULE SUPPORT TRADE STUDY

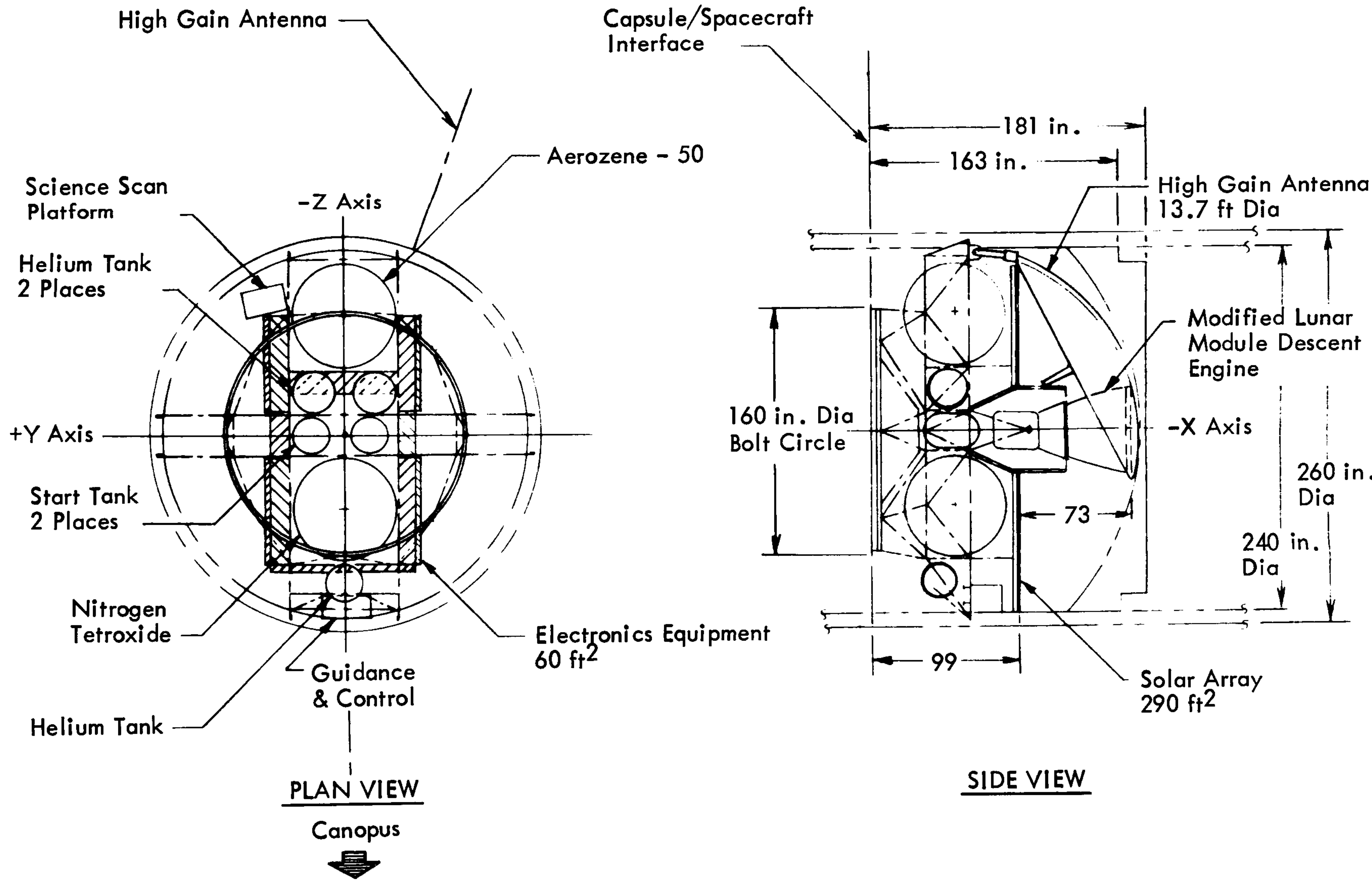


Figure 1.1.7-11: TWO-TANK CONFIGURATION

solar panel is also shown assembled to the electronics module and is not dependent on the engine installation sequence.

Semimonocoque Spacecraft Configuration -- To investigate the use of structural elements in the dual role of structural support and meteoroid protection, a semimonocoque spacecraft configuration was designed (see Figure 1.1.7-12). This configuration provides a continuous ring support at the booster shroud. Boost loads are carried from the shroud to the capsule support ring by an externally stiffened semimonocoque conical surface which is also the second plate of the meteoroid protection system. The bumper surface is attached to the outer legs of the stiffeners.

The propulsion system is supported by an inner semimonocoque cone that extends from the capsule support ring to the orbit injection engine gimbal plane. The propulsion system is removable as a unit by unbolting a splice in the outer conical surface. The propulsion system cone provides support for four main propellant tanks with internal start tanks, four propellant pressurant tanks, and the engine.

The external cone is used to support the attitude control system, electronic bays, guidance and control bays, and deployable equipment such as the science scan platform and the low, medium, and high gain antennas. The attitude control system tanks and plumbing mount on the inner surface of the outer cone. The nozzles are mounted on the outer circumference of the outer cone on the Y and Z axes. The electronic bays and the guidance and control bays are provided with upper and lower cable trays and local mounting plates and temperature control louvers. The special guidance and control bay is used to align the Canopus tracker with the +Z axis.

1.1.7.5 Configuration Comparison

The baseline and two alternate configurations are compared in Table 1.1.7-2. The four-tank configuration identified as the baseline was selected over the semimonocoque because of the weight advantage of the truss structure. The baseline configuration was selected over the two-tank configuration because it allows removal of the propulsion module as a unit.

1.1.8 Planetary Quarantine

This section describes the techniques for complying with the planetary quarantine probability apportionments listed in Figure 1.1.8-1. The apportionments were designed to guarantee that the biological contamination of the planet Mars will not occur within a probability (P_c) of 1×10^{-3}. The following procedures are proposed to ensure the probability allocation (shown in parenthesis) for each contaminating event is not exceeded:

Flight Spacecraft Accidental Impact (1×10^{-5}) -- At each midcourse maneuver, the vehicle aim points will be biased so that the probability of accidental impact with the planet Mars will be less than 1×10^{-5}.

Orbit Insertion at Encounter (2×10^{-6}) -- The orbit insertion command will not be given unless the resulting orbit satisfies this constraint.

Deorbit Decay Premature Impact (8×10^{-6}) -- The orbit shall be selected to satisfy this constraint for a period of 11 years or until after the year 1984.

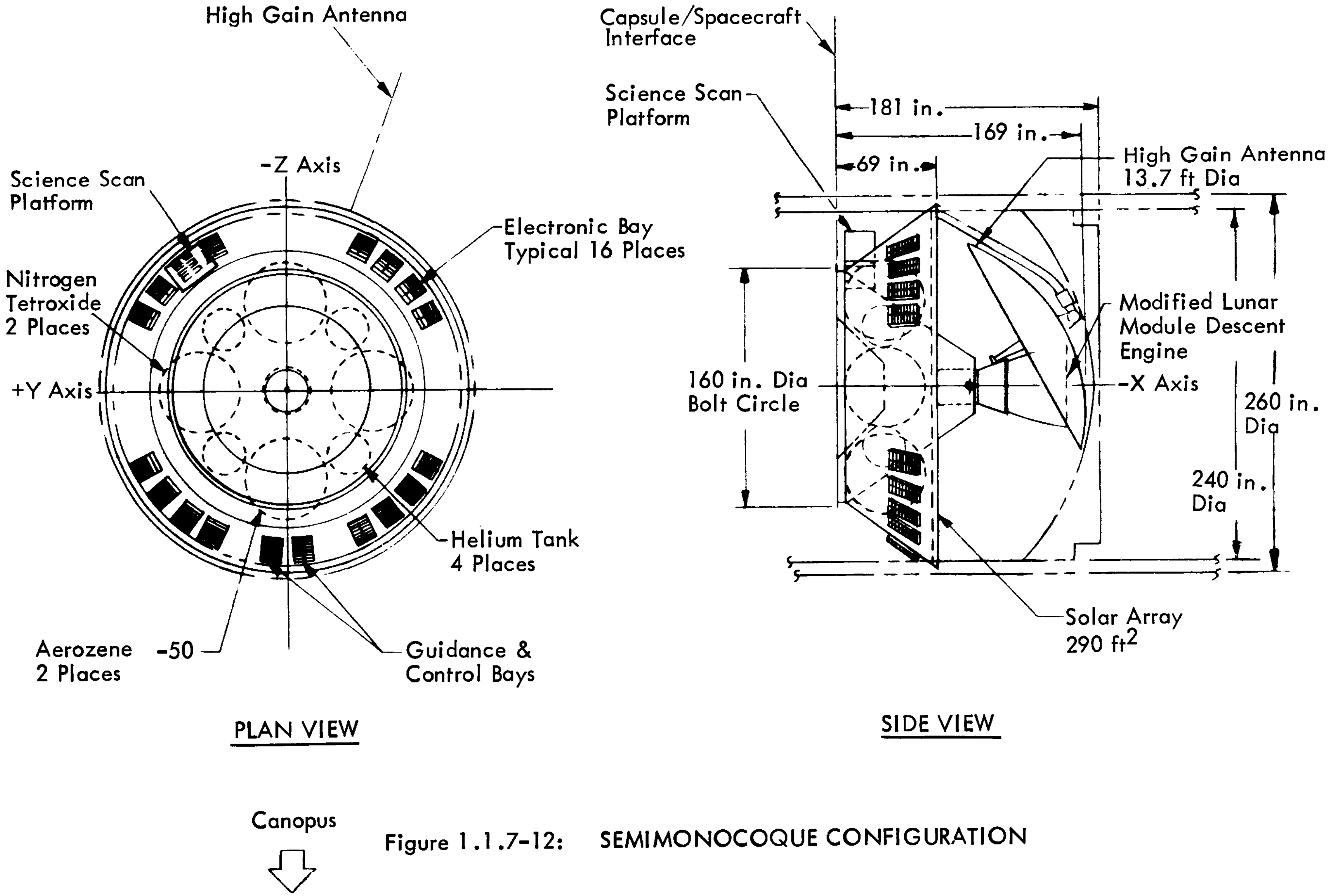


Figure 1.1.7-12: SEMIMONOCOQUE CONFIGURATION

Table 1.1.7-2: CONFIGURATION COMPARISON

	CONFIGURATION		
	BASELINE	**SEMIMONOCOQUE**	TWO-TANK
Main Propellant Tankage	4 Tanks	4 Tanks	2 Tanks
Propellant Starting System	2 Separate Start Tanks	Integral Start and Maneuver Tanks	Separate Start Tanks
Propulsion System Removal	Removable	Removable	Not Removable
Structure Type	Truss	Semimonocoque	Truss
Adapter	Integral	Integral	Integral
Planetary Vehicle Weight - 6000-lb Capsule	22,520	23,290	22,010

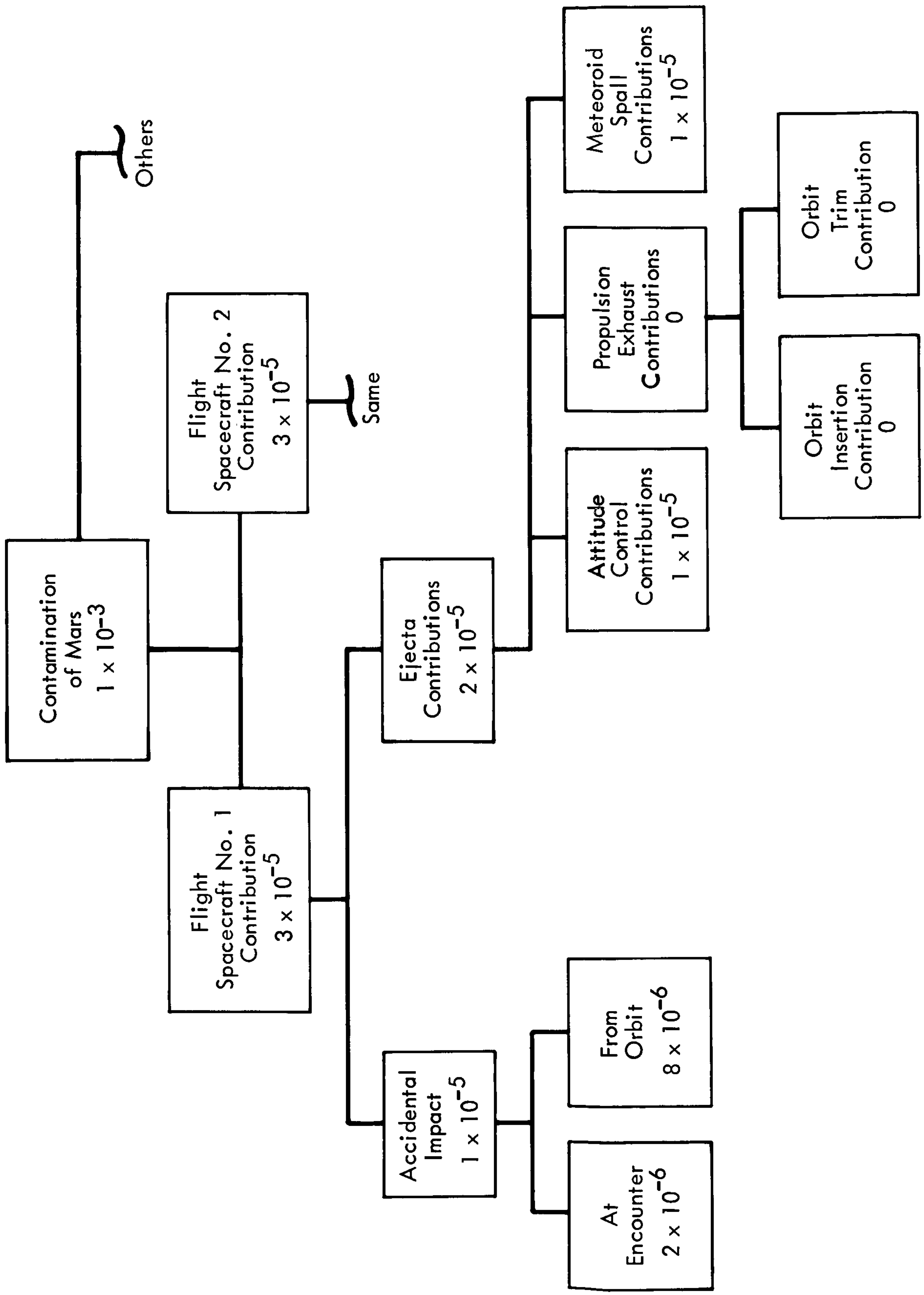


Figure 1.1.8-1: PLANETARY QUARANTINE DISTRIBUTION OF PROBABILITY OF CONTAMINATION

Ejecta Contributions (2×10^{-5}) -- Attitude control exhaust, propulsion exhaust, and ejecta as a result of meteoroid spall shall be controlled as follows to satisfy this constraint:

1) Attitude Control Ejecta (1×10^{-5}) -- The attitude control hardware shall be cleaned in accordance with the particulate contamination requirements listed in Section 1.1.9, sealed, and maintained in this condition throughout assembly, checkout, and launch. The cold nitrogen gas shall be filtered and loaded aseptically.

2) Propulsion Exhaust Ejecta (0) -- The liquid propellants, monomethylhydrazine and nitrogen tetroxide, are sporicidal. The system is expected to be fired at least two times during interplanetary flight which should flush out the system completely. Therefore, it is anticipated that, at orbit insertion and orbit trim, the probability of contamination from these sources will be zero.

3) Meteoroid Spall Ejecta (1×10^{-5}) -- Analysis indicates that biological contamination of Mars (resulting from ejecta from the spacecraft external surfaces and appendages being subjected to meteoroid impact) can be controlled within the above constraint by close control of the original microorganism loading. Assembly of the spacecraft in a Class 100,000 clean room, per Federal Standard Number 209a, will keep the original microbial loading conservatively below 1×10^7 for the complete flight spacecraft. This will be reduced to 10^5 in Mars orbit by effect of ultraviolet impingement, meteoroid impact during interplanetary cruise, and the dieoff of the organisms during the complete mission. With the microbial loading on the exterior surfaces reduced to 10^5 in Mars orbit, the probability of contaminating Mars from meteoroid impact ejecta is less than 1×10^{-5}. Accordingly, decontamination of the external surfaces of the flight spacecraft will not be required, provided adequate care is exercised to keep the original microbial loading down.

In-Transit Ejected Hardware (0) -- All hardware separated from the launch vehicle and the flight spacecraft shall be released in a manner and at a time that will place it on a Mars nonimpact trajectory.

1.1.9 Particulate Contamination

The key problem related to particulate contamination is associated with the Voyager spacecraft operational reliability. Particulate contamination can cause malfunctions and/or false signals to the optical sensors and science subsystems, malfunctions in hydraulic and pneumatic systems, short circuits and circuit characteristic changes in electronic systems, and changes in the emittance and absorptance of thermal control coatings. To prevent these malfunctions, the following procedures will be used:

1) Computing and Sequencing, Telecommunications, Cabling, Power, Thermal Control Science Pyrotechnic Subsystems -- Individual components will be received from point of manufacture in a cleaned condition and double-bagged. When it is necessary to open the bag and inspect or test the item, it will be accomplished in a Federal Standard 209a, Class 100,000 clean room. Individual operations that require a higher degree of cleanliness will be accomplished

in laminar flow clean benches rated by Federal Standard 209a as Class 100. Structural components (containers for electronic subsystems) will be fabricated in normal shop environments and then cleaned, painted, and moved into a controlled cleaning facility where the part will be given a final cleaning and packaging following procedures established for Lunar Orbiter. If it becomes necessary to remove the part from the package, it will be done in a Class 100,000 clean room. Should it be necessary to remove the part from the clean room for an unplanned operation in a noncontrolled environment, the part will go back through the cleaning and packaging operation prior to being readmitted to the clean room. Soldering and welding of electronic parts will be accomplished within the areas that meet or exceed the requirements established by NASA Quality Publication NPC 200-4, August 1964, Section 2. The knowledge gained on Lunar Orbiter operations indicates that frequent cleaning, wiping, and/or vacuuming of parts is necessary to maintain a high level of parts cleanliness. These cleaning tasks will be written into approved procedures, and strict compliance with the procedures will be stressed throughout the assembly and test operation.

2) Propulsion and Attitude Control Subsystems -- Items in these subsystems will be treated the same as in paragraph 1) above, except where tubing, valves, regulators, and other plumbing-type items are involved. The internal cleanliness will be achieved by a cleaning and flushing operation performed within an area meeting Class 100 particulate limitations. Cleanliness levels of interior surfaces are verified by measuring the contaminates deposited by the flushing solution on a millipore filter. The cleaning, flushing, checking, and packaging operation will be accomplished within a Class 100 environment. The assembly (brazing) of these supercleaned parts will take place in a Class 100,000 clean room, within a Class 100 clean bench. All fluids or gases used in subsequent testing will be cleaned to meet the required level of cleanliness, i.e., no metallic particles greater than 5 microns and no nonmetallic particles greater than 25 microns. Once the systems have been assembled and sealed, they will not be opened except in an appropriate environment. The exterior of the systems can be exposed to noncontrolled environment without being detrimental to the subsequent operation of the system. However, external cleaning procedures (wiping and vacuum) will be used after such exposure and prior to admittance to a controlled area.

3) Structural and Mechanical Subsystems -- These items will be fabricated and preassembled in a normal factory area using the normal, high quality space hardware housekeeping standards. When the prefit operation has been completed, the parts will be cleaned, painted, and then cleaned and packaged for clean room assembly. After the subsystems have been assembled into the spacecraft in a Class 100,000 clean room, the completed unit will be moved between controlled locations only after the spacecraft has been properly sealed with a protective barrier that will not be removed until the spacecraft is within the next controlled area.

1.2 FUNCTIONAL DESCRIPTION FOR SPACECRAFT HARDWARE SUBSYSTEMS

1.2 FUNCTIONAL DESCRIPTION FOR SPACECRAFT HARDWARE SUBSYSTEMS

Each of the spacecraft subsystems has been described in terms of design constraints and requirements; functions; performance and physical characteristics; interfaces; and reliability. Summaries of trade studies performed during Task D, for subsystem selection, are included. New technology and development, if required, are defined and scheduled for each subsystem. The growth potential of each subsystem is discussed.

1.2.1 Power Subsystem

1.2.1.1 Design Constraints and Requirements
1.2.1.2 Functional Description and Performance Characteristics
1.2.1.3 Physical Characteristics
1.2.1.4 Interface Definition
1.2.1.5 Reliability
1.2.1.6 Trade Study Summary
1.2.1.7 New Technology and Development Items
1.2.1.8 Growth Potential

1.2.1 Power Subsystem

The power subsystem supplies the spacecraft electrical power during all Voyager flight regimes. Solar cells convert solar radiation to electric power when the spacecraft is Sun-oriented and the solar array is illuminated. Power is supplied from a set of storage batteries when the spacecraft is occulted from the Sun.

The solar cells are interconnected into series and parallel groups that supply power at a voltage compatible with the load and power conditioning requirements. The solar cell groups are mounted on a set of flat panels making up the solar array. The solar array is oriented normal to the Sun vector within ±0.3 degree (under normal conditions) by the attitude control subsystem.

The spacecraft storage batteries are recharged from the solar array after spacecraft reference acquisition, after each off-Sun maneuver, and after each period when the spacecraft is Sun-occulted.

Regulated, 50-volt rms, 2400-Hz, single-phase power is supplied to all electronic loads other than the traveling wave tube amplifier (TWTA) from a set of regulated inverters. The inverters receive d.c. from either the solar array or batteries.

1.2.1.1 Design Constraints and Requirements

The power subsystem requirements and constraints appear in Table 1.2.1-1. The raw power requirements as a function of mission regime are shown in Figure 1.2.1-1.

The flight regime that sizes the solar array is labeled "S/C Orbit Recharge I" in Figure 1.2.1-1. The power demand during this regime is second highest and occurs in Mars orbit when the solar intensity, hence, array power capability, is lowest. The regime labeled Cruise II requires 15% more power, but occurs at a time when the solar array power capability is, due to higher solar intensity, over 20% greater than in Mars orbit.

The power demand following the third midcourse correction (labeled Cruise III in Figure 1.2.1-1), when the solar intensity may only be 4% above that encountered in Mars orbit, is reduced by 200 watts from the Cruise II load by turning off capsule power for the duration of battery recharge.

The load demand for each subsystem during the critical regime that sizes the solar array is shown in Table 1.2.1-2.

Table 1.2.1-1: DESIGN CONSTRAINTS AND REQUIREMENTS — 1973 Mars Mission

Maximum Sun-Mars Distance	1.66 A.U.
Maximum Sun-Mars Distance at Mars Arrival	1.63 A.U.
Maximum Orbital Period	12.4 hours
Occultation, First 30 Days	None
Maximum Sun Occultation Period	1.5 hours
Minimum Spacecraft Life in Orbit	6 months
Maximum Capsule Power	200 watts
Capsule Support Power	20 watts
Solar Array Configuration	Fixed
Largest Solar Array Diameter	240 inches
Solar Array Temperature Range: Operating	-145°C to +115°C
Nonoperating	-145°C to +150°C
Minimum Reliability	0.991

Table 1.2.1-2: CRITICAL REGIME LOAD DEMAND — Spacecraft Orbit - Recharge I

AC LOADS	WATTS
Computing and Sequencing	68.8
Pyrotechnics	11.8
Guidance and Control	64.6
Radio	23.0
Antenna Drive	3.0
Capsule Support	20.0
Telemetry	16.9
Data Storage	54.0
Propulsion Electronics	9.0
Science	45.0
Subsystem Monitor	10.0
TOTAL AC LOAD	326.1
Regulated Inverter Losses	110.5
DC LOADS	
Capsule	0.0
TWT Amplifier	200.0
Propulsion Solenoids	0.0
Reaction Control Jets	0.0
TVC Actuator	0.0
Battery Chargers	239.0
TOTAL DC LOAD	439.0
Power Switching Logic Losses	28.6
SOURCE POWER REQUIRED (DC)	904.2

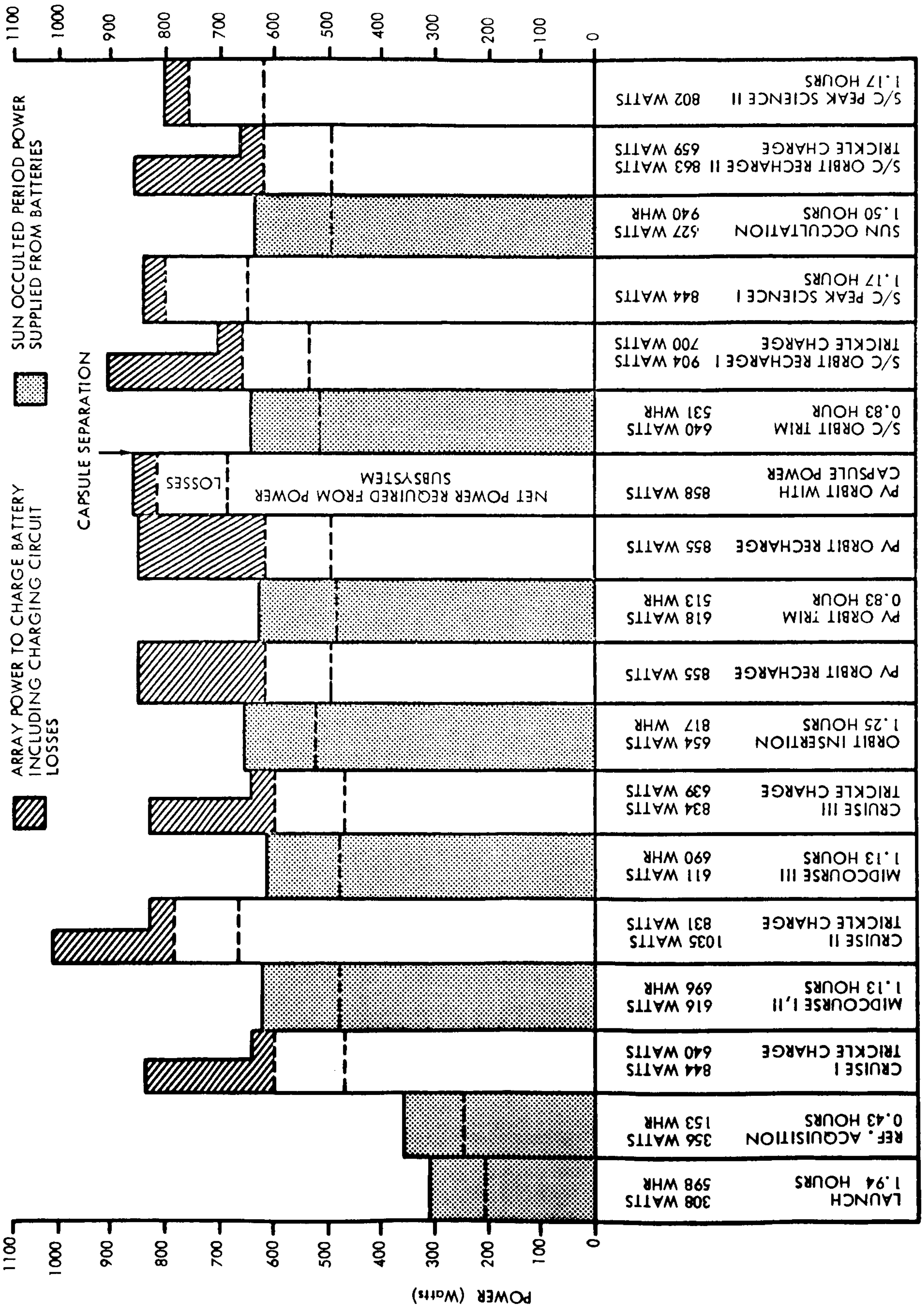


Figure 1.2.1-1: SOURCE POWER REQUIREMENTS

NOTE: Short duration peaks not shown.

1.2.1.2 Functional Description and Performance Characteristics

Electrical power is generated in a 16-panel solar array and is stored in three sealed nickel-cadmium (Ni-Cd) batteries. Power to all electronic and science loads other than the TWTA is distributed through three separate buses at 50 volts regulated a.c. and 2400-Hz frequency. The loads are so connected to these buses that the mission can be completed successfully even if power on one bus is lost.

There are two TWTA's, but only one is on at any time. Each TWTA is supplied through its own feeder from an unregulated d.c. bus (Figure 1.2.1-2). This bus receives power from the solar array whenever the array is illuminated. Diodes automatically connect the three batteries to this bus at other times.

During spacecraft maneuvers, power is supplied to propulsion and guidance-and-control solenoid valves from the three batteries through the maneuver bus. Power to the capsule is supplied only from the solar array through the capsule bus and dual feeders. Capsule power is not supplied when the solar array is not illuminated, as during spacecraft midcourse corrections and orbit insertion. Capsule power is also turned off during battery recharge following Mars orbit insertion.

Each solar panel and each battery is connected to the unregulated d.c. bus through a diode. A short circuit in any one panel or battery is thus isolated from the unregulated bus.

The rate of recharge of each battery, following a discharge period, is controlled by a battery charger. Each charger is supplied from two solar panels through diodes. Thus, a battery-charger fault that is not neutralized within the battery charger will affect only 2 of 16 solar panels.

The unregulated d.c. power obtained from the solar array or battery is regulated and inverted to 50-volt rms, 2400-Hz square-wave a.c. power with three separate inverters, each supplying one of the three a.c. buses. Each inverter has a malfunction detector that disconnects the inverter from the unregulated d.c. bus if the input current becomes excessive or if the output voltage goes out of tolerance.

A power subsystem monitor provides power for malfunction detection within the inverters and battery chargers. It conditions the command signals and it also provides excitation and conditioning for all voltage, current, and temperature-sensing required for control and telemetry. The monitor derives its input power from all three a.c. buses. Reactive current-limiting is used to protect the buses from a monitor malfunction. A share sensor and boost converter is provided to prevent the batteries from sharing in supplying power to the spacecraft following the decay of power peaks that exceed the solar array capability.

The performance characteristics of the key elements of the power subsystem are described in the following paragraphs:

1) Solar Array -- The solar array consists of 16 pie-shaped solar panels having a total area of 290 square feet on which 60,480 solar cells are mounted. The selected solar cells are 2 cm by 2 cm, 12 mils thick, with 2 ohm-cm resistivity, and have an efficiency of 11% based on their active cell area. The solar cells are protected by 6-mil cover glasses.

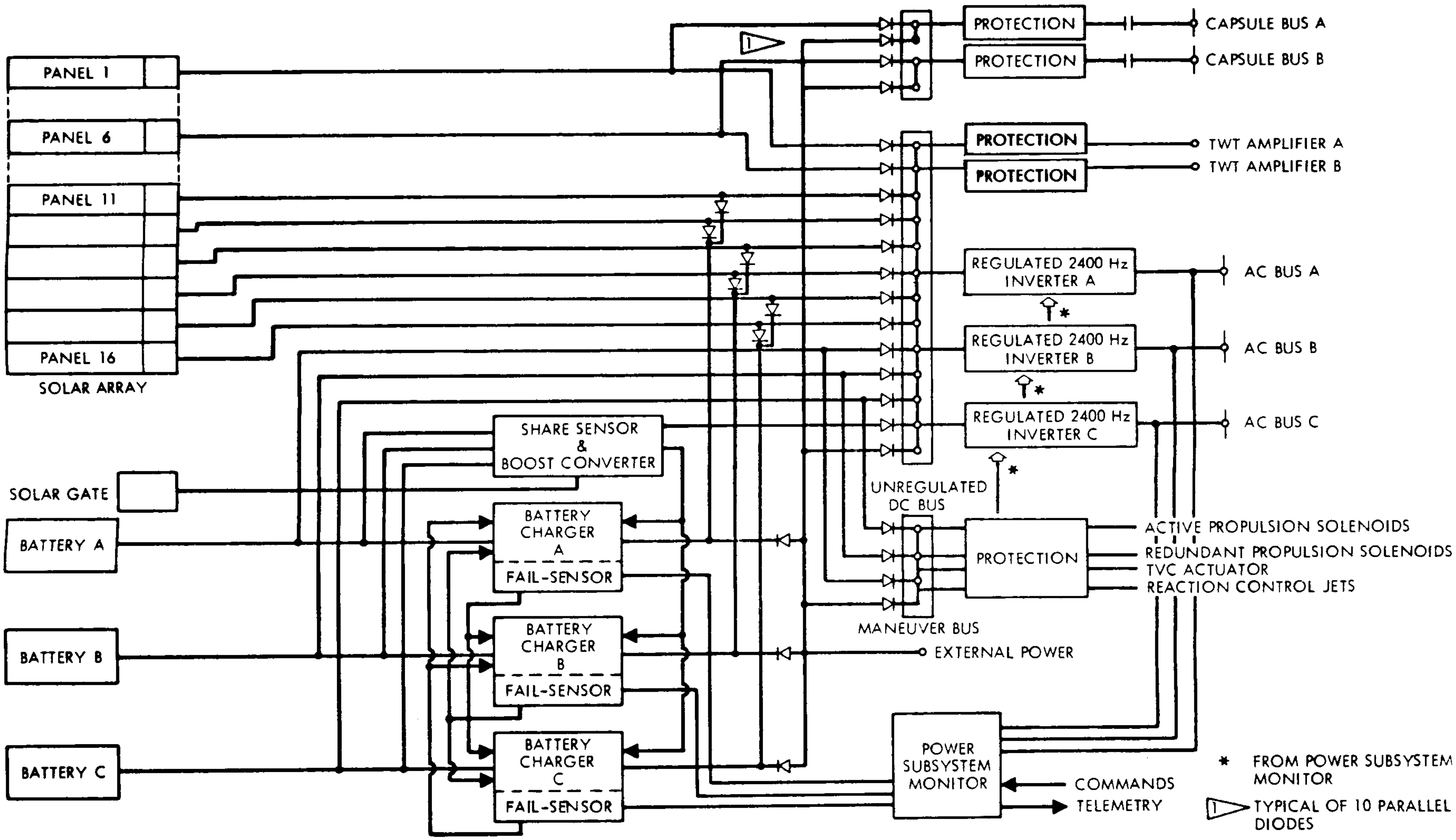


Figure 1.2.1-2: POWER SUBSYSTEM BLOCK DIAGRAM

The total radiation seen by the solar cells during the 1973 Voyager Mars mission is estimated to be less than that from a single equivalent solar flare having a fluence of 10^9 protons per square centimeter. The corresponding 1-Mev electron flux is 4×10^{12} electrons per square centimeter. At this fluence, the radiation degradation of 2 and 10 ohm-cm solar cells with a 6 mil cover glass is 0 to 1%. A 2 ohm-cm cell was chosen over a 10 ohm-cm cell to take advantage of its 4% higher power output under nominal conditions. The solar array design factors are shown in Table 1.2.1-3.

Table 1.2.1-3: SOLAR ARRAY DESIGN FACTORS

FACTOR	VALUE
Cell Efficiency at Air-Mass Zero	0.11
RMS Measurement Uncertainty	0.958
Cover Glass Loss	0.95
Process and Mismatch Loss	0.97
Thermal Cycling Degradation	0.95
Micrometeorite Degradation	0.99
Radiation Degradation	0.99
Low Illumination Loss	0.98
Solar Constant Uncertainty	0.98
Array Packing Factor	0.90
Operation Off the Maximum Power Point	0.95
Wiring Loss	0.98
Active Cell Area/Total Cell Area	0.95

The solar array view to space for radiation of heat is partly obscured by the engine in front and the spacecraft in the rear. The temperature difference between the array inner edge and outer edge is estimated to be 47°C with a median temperature of 80°C when near the Earth and of 0°C in Mars orbit. (See Figure 1.2.1-3). During a 1.5-hour Sun occultation at Mars, the median solar array temperature will drop to as low as -133°C. The panel will recover to a 0 ± 10°C temperature within 0.5 hour after it is illuminated with sunlight.

The solar array must operate at or above 62 volts to charge fully a 38-cell Ni-Cd battery at 0°C. The array voltage, at a given load, is the lowest in the vicinity of Earth when the solar cells are at 80°C. A solar cell string of 135 series-connected cells develops the required 62 volts. However, this array will produce a voltage of 109 volts (which puts 108 volts on the unregulated bus) when the array is illuminated at -133°C. This maximum voltage condition occurs after the end of Sun occultation at Mars. The voltage drops to approximately 80 volts within 0.5 hour as the solar array temperature rises to 0°C.

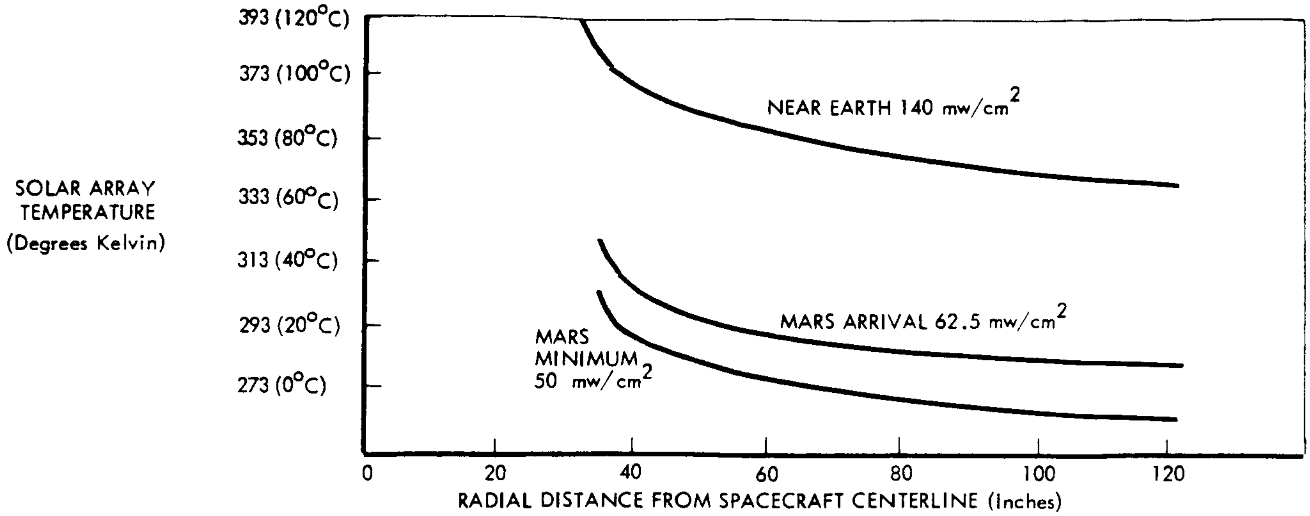


Figure 1.2.1-3: SOLAR ARRAY TEMPERATURE PROFILES

The volt-ampere characteristics of the array under near-Earth, Mars steady-state, and Mars sunrise conditions are shown in Figure 1.2.1-4.

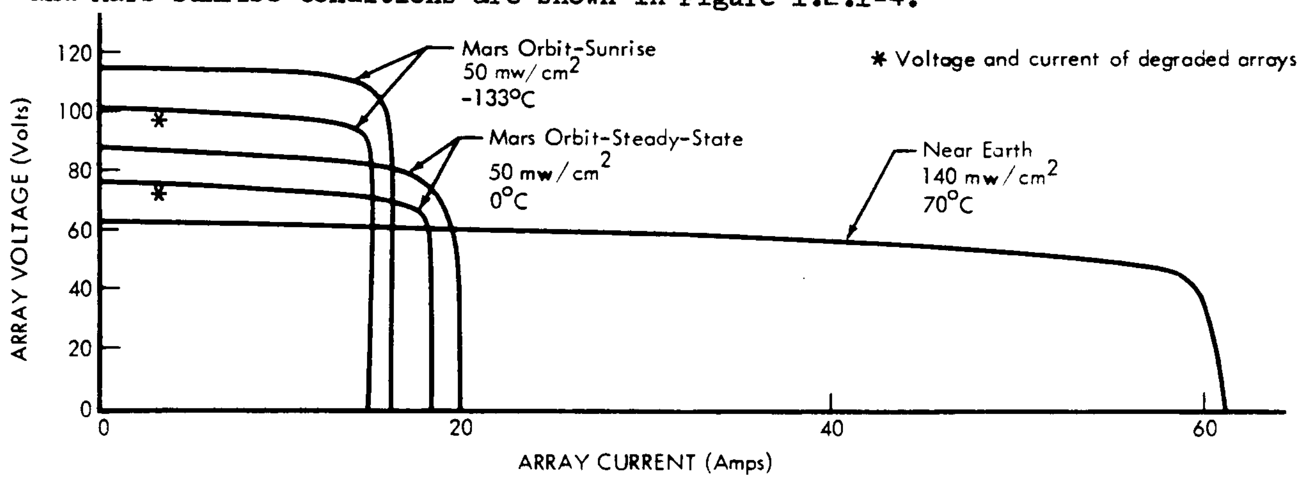


Figure 1.2.1-4: SOLAR ARRAY VOLT-AMPERE CHARACTERISTICS

2) Battery -- The battery is sized to supply the power required for the Sun-occulted regimes in Figure 1.2.1-1. The minimum battery voltage is 38 volts, and up to 500 charge-discharge cycles may be required. A battery composed of 38 series-connected, 20 ampere-hour, Ni-Cd cells operating at a cell temperature of 10 ± 10°C is capable of meeting the voltage, power, energy, and life requirements. The depths of discharge for two and three batteries operating in parallel are shown in Table 1.2.1-4.

3) Power Conditioning -- Some loads are composed of redundant elements that can be supplied from separate power buses to achieve overall redundancy by means of parrallel multichannel design. Examples are the computing and sequencing logic unit and a portion of the guidance and control. Other loads are provided with the capability of transferring from a normal bus to an alternate bus in the event that power at the normal bus fails. The inverter rating is established by the maximum load that is connected to, or can be transferred to, its bus. Thus, each inverter must be capable of carrying a 210-watt load (Table 1.2.1-5).

Table 1.2.1-4: BATTERY DEPTH OF DISCHARGE

FLIGHT REGIME	WATT-HOURS REQUIRED	PERCENT DEPTH OF DISCHARGE	
		2 OF 3 BATTERIES	3 BATTERIES
Launch and Reference Acquisition	751	41%	27%
Midcourse Correction	696	38	25
Orbit Insertion	817	45	30
Planetary Vehicle Orbit Trim	513	28	19
Spacecraft Orbit Trim	531	29	19
Spacecraft Orbit Occultation	940	52	34
Based on 38 Series-Connected, 20 amp-hr Cells Per Battery Operating at 1.2 Volts/Cell			

Table 1.2.1-5: SUBSYSTEM ELECTRICAL LOAD DISTRIBUTION
(Alternating Current — Power in Watts)

SUBSYSTEM	NORMAL OPERATION			BUS A FAILED		BUS B FAILED		BUS C FAILED	
	BUS A	BUS B	BUS C	BUS B	BUS C	BUS A	BUS C	BUS A	BUS B
Computing & Sequencing	38.8	---	30.0	---	30.0	38.8	30.0	38.8	4.0
Pyrotechnics	5.9	---	5.9	---	5.9	5.9	5.9	5.9	---
Attitude Ref. Gyros	25.0	25.0	---	25.0	25.0	25.0	25.0	25.0	25.0
Attitude Ref. Electronics	---	0.6	4.0	0.6	4.0	---	4.0	---	0.6
Autopilot	---	5.0	5.0	5.0	5.0	---	5.0	---	5.0
Radio	---	10.4	12.6	10.4	12.6	---	23.0	---	23.0
Antenna	---	---	3.0	---	3.0	---	3.0	---	3.0
Telemetry	---	---	16.9	---	16.9	---	16.9	16.9	---
Data Storage	24.0	---	30.0	---	54.0	24.0	30.0	54.0	---
Propulsion	---	4.5	4.5	4.5	4.5	---	4.5	---	4.5
Science	49.0	92.0	22.0	121.0	22.0	101.0	42.0	49.0	114.0
Capsule Support	11.0	9.0	---	9.0	11.0	11.0	9.0	11.0	9.0
Power Subsystem Monitor	3.3	3.3	3.3	5.0	5.0	5.0	5.0	5.0	5.0
Total	157.0	149.8	137.2	193.3	198.9	210.7	203.3	205.6	193.1

Providing power to spacecraft loads at a regulated a.c. voltage minimizes the need for additional regulation downstream of the power subsystem interface. The regulation band is +2%/-3%. The regulated inverter output characteristics are:

Voltage - Square-wave, 50 volts rms having a rise time of 5 ± 4 microseconds.

Frequency - 2400 Hz ± 0.01% with synchronization and ±5% without synchronization.

An external square-wave synchronizing signal is provided to each inverter from the computing and sequencing subsystem to control the inverter frequency. In the absence of this signal, the inverter derives its frequency control from an internal magnetic oscillator. The inverter efficiency is between 75 and 81% at rated load with 37 to 108 volts d.c. input. The inverter is rated at 230 watts, providing a 10% design margin, and is capable of supplying a peak current of 150% of rating for 10 milliseconds.

The output voltage and input current of the regulated inverter are monitored by a fail sensor. An overvoltage, or an undervoltage deviating more than 10% from nominal output voltage, or an overcurrent exceeding two times the maximum input current will cause the fail-sense circuit to disconnect the regulating inverter from its power source. The inverter can also be switched on and off by Earth command.

The fail-sense circuit of each inverter is supplied 28 volts d.c. power from the power subsystem monitor to ensure that the fail-sensing function is not affected by failure within the regulating inverter.

4) Battery Recharge and Control -- The batteries are discharged to a depth indicated in Table 1.2.1-4, during launch, reference acquisition, maneuvers, and Sun occultation in Mars orbit. Energy is restored to the battery from the solar array. A separate battery charger controls the rate of recharge of each battery.

A constant-current charge rate of 1.14 amperes was selected to minimize the power demand from the solar array, yet fully charge the battery prior to the next Sun occultation when in Mars orbit. The 1.14-ampere charge duration is based on restoring 1.6 times the ampere-hours previously discharged as measured with a coulometer. The 1.14-ampere charge is followed by 0.2-ampere trickle charge. Switchover to the trickle charge rate will also occur if the monitored battery voltage exceeds 58.9 ± 0.4 volts d.c. at 275°K or a lower voltage at higher temperatures.

A fail sensor within the charger will disconnect the charger from the solar array should the battery temperature exceed 325 ± 2°K, or should the battery voltage fall below 39 volts during charging. The power subsystem monitor supplies 28 volts d.c. power to each charger for failure signaling and clearing. For example, a disconnected charger will signal the other two chargers to increase their charge rates by 50% during the normal charge mode. The signal interconnections between battery chargers are shown in Figure 1.2.1-2.

5) Power Switching and Logic Unit -- The solar panels and batteries are connected to the unregulated bus through diodes, as shown in Figure 1.2.1-2. The diode connecting each battery to the unregulated bus serves to isolate the bus and the other two batteries from a short circuit in a battery or in its associated battery charger. The diode arrangement also prevents a short-circuited battery charger from disabling more than two of the 16 solar panels. Because of its magnitude (200 watts), capsule power is supplied only from the solar array and not from the batteries. The capsule bus is isolated from the unregulated d.c. bus, and hence the batteries, by connecting the capsule bus to 10 of the 16 solar panels through 10 separate diodes as shown in Figure 1.2.1-2.

The propulsion solenoids, TVC actuators, and reaction control jets are supplied from all three batteries through diodes and the maneuver bus. The resultant isolation between the unregulated d.c. bus and the maneuver bus reduces the maximum voltage at the maneuver bus to 60 volts as compared to 108 volts estimated for the unregulated d.c. bus. External power is supplied to the capsule, maneuver, and unregulated d.c. bus, and to the three battery chargers, through isolating diodes.

6) Power Subsystem Monitoring and Protection -- Control power to the battery charger and inverter fail-sensors is supplied from the power subsystem monitor. The control voltage of 28 volts d.c. is compatible with qualified relay designs. The monitor power comes from the three a.c. buses. Reactive current limiting is used to protect the a.c. buses from a malfunction within the monitor. This 2400-Hz power is also used as a source of a.c. excitation for all magnetic current sensors required for telemetry and protection.

Space-qualified latching-type relays triggered from magnetic overcurrent sensors are tentatively selected for protection of the three d.c. buses.

The power subsystem monitor processes the command signal to turn capsule power on and off during battery recharge. The monitor is estimated to consume 10 watts of power.

7) Elimination of Battery Sharing Mode -- Inrush current to the science or capsule load may result in a total power demand in excess of the solar array capability. The unregulated bus voltage then drops to a level at which the batteries share in supplying power to the spacecraft loads. The constant-power input characteristic of the regulated inverters can cause the batteries to continue to share in supplying the spacecraft load even after the power demand has declined to as low as 15% below the maximum solar array power capability. The share sensor recognizes the resulting lower operating voltage and triggers the boost converter to momentarily boost the power subsystem voltage, using battery energy. This power pulse shifts the subsystem voltage to a stable level above the array maximum power voltage.

A solar gate, composed of a set of solar cells, signals the share sensor that the array is illuminated. The share sensor functions only when the solar array is illuminated.

8) <u>AC Load Distribution</u> -- The electronic loads are distributed among the three a.c. buses, as shown in Table 1.2.1-5. Redundant loads such as computing and sequencing and gyros are connected to different buses. Nonredundant loads have relays for transferring them to another bus in case of failure of their normal a.c. bus. The loads are so distributed as to nearly balance the power demand between the remaining two inverters when any one of the three inverters has failed. The degree of balance is shown in Table 1.2.1-5. The efficiency of the inverter at 50% of rated load is approximately 6% lower than at rated load. Balancing of the loads not only reduces the inverter rating, but permits more efficient operation under normal conditions.

1.2.1.3 Physical Characteristics

The power subsystem occupies two equipment bays. The batteries and battery chargers are placed in the same bay to facilitate regulation of battery temperature at 10 $\pm$ 10°C. The subsystem component dimensions, volume, and weight are shown in Table 1.2.1-6.

Table 1.2.1-6: POWER SUBSYSTEM PHYSICAL CHARACTERISTICS

UNIT	WEIGHT EACH (Pounds)	SIZE (Inches)	VOLUME EACH (Cu In)	NUMBER	TOTAL WEIGHT (Pounds)	TOTAL VOLUME (Cu In)
Diode Logic & Switching	6.3	5 - 1/2 x 16 x 6	530	1	6.3	530
Booster Converter	2.1	5 - 1/2 x 16 x 1 - 1/2	132	1	2.1	132
Protection	3.1	5 - 1/2 x 16 x 1 - 1/2	132	1	3.1	132
Regulated Inverter	12.8	5 - 1/2 x 16 x 6	530	3	38.4	1590
Battery Charger	4.6	5 - 1/2 x 16 x 1-1/2	132	3	13.8	396
Subsystem Monitor	1.0		10	1	1.0	10
Miscellaneous	9.3				9.3	
Subtotal					74.0	
Battery (Ni-Cd)	83.0	8 x 22 1/2 x 8 1/2	1530	3	249.0	4590
Subtotal					323.0	
Solar Array (290 ft^2)					250.0	
Total Weight					573.0	

1.2.1.4 Interface Definition

The interfaces between the power subsystem and the spacecraft and its other subsystems are defined in Table 1.2.1-7.

1.2.1.5 Reliability

The assessed reliability for the power subsystem in 0.989. This essentially meets the goal of 0.991 established for this subsystem. The solar array reliability assessment was obtained by extrapolation of previous analyses on the basis that the power margin for the array will be the same. Reliability assessment for the subsystem elements are provided in Table 1.2.1-8.

Table 1.2.1-7: SUBSYSTEM INTERFACES

ITEM NO.	TYPE OF INTERFACE	INTERFACE DESCRIPTION	INTERFACING SUBSYSTEM	INPUT		BOUNDARY DEFINITION
				TO	FROM	
1	Electrical	Power Subsystem Power Output - Bus A : 50 v rms, 2400 Hz, ac	Computing & Sequencing Pyrotechnic Guidance & Control Data Storage Science Capsule Support	X		Bus A cable connection at 2400 Hz inverter
2	Electrical	Power Subsystem Power Output - Bus B: 50 v rms, 2400 Hz, ac	Guidance & Control Radio Propulsion Science Capsule Support	X		Bus B cable connection at 2400 Hz inverter
3	Electrical	Power Subsystem Power Output - Bus C: 50 v rms, 2400 Hz, ac	Computing & Sequencing Pyrotechnics Guidance & Control Radio Telemetry Antenna Data Storage Propulsion Science	X		Bus C cable connection at 2400 Hz inverter
4	Electrical	TWT Amplifier Power, 37-108 v dc (0-200 watts)	Radio	X		2 cable connections at main dc bus
5	Electrical	Propulsion Solenoids Power, 37-60 v dc (0-675 watt pulses)	Propulsion	X		2 cable connections at maneuver bus
6	Electrical	TVC Actuator Power, 37-60 v dc (0-300 watt pulses)	Guidance & Control	X		2 cable connections at maneuver bus
7	Electrical	Reaction Control Jet Power, 37-60 v dc (0-60 watt pulses)	Guidance & Control	X		2 cable connections at maneuver bus
8	Electrical	Capsule Power, 63-108 v dc (0-200 watts)	Capsule	X		2 cable connections at capsule bus
9	Electrical	Power Subsystem Telemetry Signals (0-5 v dc) (40 measurements)	Telemetry	X		Cable connection at Power Subsystem Monitor
10	Electrical	Power Subsystem Command Signals (23 commands)	Computing & Sequencing		X	Cable connection at Power Subsystem Monitor, Regulated Inverter, and Battery Charger
11	Thermal Design	Control power subsystem temperatures throughout all ground test and mission phases	Thermal Control		X	Component mounting surfaces
12	Thermal	Heat radiation from solar array	Spacecraft	X		Space
13	Thermal	Heat conduction from solar array	Spacecraft	X		Thermal barrier
14	Structural	Mounting provisions for power subsystem cases	Structural & Mechanical		X	Mounting hole pattern
15	Solar Radiation	Solar Array - Solar radiation converted to electrical power	Power	X		Solar Cell
16	Electrical	External Launch Complex power, 37-110 v dc, 600 watts	Launch Complex - Power		X	Umbilical Connector
17	Electrical	Launch Complex control and switching	Launch Complex - Equipment			Umbilical Connector
		a. Battery monitoring signals		X		
		b. Int/Ext power relay monitor signals		X		
		c. Int/Ext power relay control signals			X	

Table 1.2.1-8: RELIABILITY ASSESSMENT

SUBSYSTEM ELEMENTS	RELIABILITY ASSESSMENT	RELIABILITY GOAL
Solar Array	0.9999	
Power Subsystem Monitor	0.9999	
Power Switching & Logic	0.9999	
Boost Conv. & Share Sensor	0.9940	
2400 Hz Power	0.9991	
(Each 2400 Hz Inverter)	(0.9832)	
Battery Power	0.9965	
(Each Battery & Charger)	(0.9653)	
TOTAL SUBSYSTEM	0.989	0.991

1.2.1.6 Trade Study Summary

Trade studies conducted during Task A and B power subsystem developments were reviewed. Those trade studies affected by the new Task D requirements were updated.

The trade studies produced the following conclusions:

Trade Study	Conclusion
Battery type	Sealed nickel-cadmium
Type of d.c. voltage regulator	Series switching
a.c. or d.c. distribution	a.c. distribution
Diode fault isolation or fail-safe array	Diode fault isolation

Figures 1.2.1-5 through 1.2.1-8 show the considerations on which these conclusions are based.

1.2.1.7 New Technology and Development Items

The battery, with its 0.967 estimated mission reliability, is the least reliable element of the power subsystem. Improvements in battery reliability will contribute to the probability of mission success, particularly in longer and more heavily loaded post-1973 flights. Control of battery overcharge is a key to obtaining longer battery life. Methods are:

1) Charge at constant current, then reduce charging current to a trickle value when voltage exceeds a specified value. This method, currently in use, compromises battery life by permitting excessive overcharge because temperature sometimes has a greater effect on battery voltage than has state of charge.

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE: BATTERY TYPE		
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMNTS		MATRIX OF DESIGN APPROACH: ALTERNATE 1 SILIVER-CADMIUM BATTERY	MATRIX OF DESIGN APPROACH: ALTERNATE 2 NICKEL-CADMIUM BATTERY	SELECTION

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMNTS

The Voyager spacecraft, when operating with one battery failed and two batteries in service, requires the following amounts of energy from the batteries at a voltage not less than 38 volts:

During launch and reference acquisition, 809 watt-hours are required. Only one charge-discharge is required, hence deep discharge (say 75%) consistent with reliability is permissible. The minimum end of discharge voltage is 37 volts.

During Mars orbital operation, up to 653 watt-hours of energy is required when the spacecraft is shadowed from the Sun by Mars. Up to 420 cycles of charge-discharge will be required. Depth of discharge shall be consistent with a mission reliability of 0.967 for one battery.

During the Earth-Mars transit there will be up to seven discharges, separated by periods of charge, in which up to 578 watt-hours will be drained from the battery during each discharge.

The duration of a charge-discharge cycle at Mars can be between 7 and 14 hours.

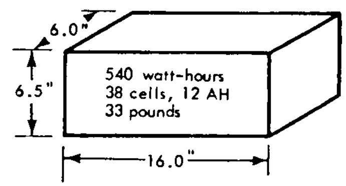
The battery will be mounted on a cold plate which will be at a temperature between 40°F and 95°F.

The normal operating life of the battery will consist of 170 to 210 days of Earth-Mars transit with little activity, followed by 180 days of orbital operation at Mars during which there will be several periods of cycling.

The battery must withstand vibration during launch and the vacuum of space thereafter.

During the 6 months prior to launch the battery may accumulate 50 charge-discharge cycles.

ALTERNATE 1 SILIVER-CADMIUM BATTERY



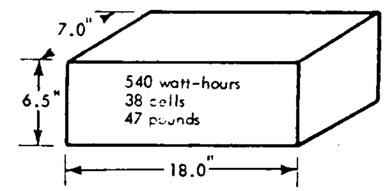
PRO

1. Lighter
2. No Magnetic Material
3. Smaller

CON

1. Development of the silver-cadmium battery is being emphasized by only one manufacturer with limited government support.
2. The battery size and weight shown is based on an undeveloped cell. The next larger existing 18-AH cell would result in a 52-pound battery.
3. Providing cell reliability can cost 9 man-years of effort.
4. Requires sophisticated charge control.

ALTERNATE 2 NICKEL-CADMIUM BATTERY



PRO

1. Requirement can be met by a 12 AH cell that has been used on the Lunar Orbiter, OGO-1, and OGO-3. The OGO-1 battery operated in space for 2-1/2 years (May, 1967) which is approximately the required life for Voyager.
2. Nickel-cadmium battery charging requires no new techniques.
3. The state of art is being advanced by substantial government funding, and extensive reliability test programs are in progress.

CON

1. Heavier
2. The resulting spacecraft will not be magnetically clean for future missions.

SELECTION

Probability of no failure during Voyager mission

Alt. 1 - 2
Alt. 2 - 1

Low weight

Alt. 1 - 1
Alt. 2 - 2

Magnetic Cleanliness

Alt. 1 - 1
Alt. 2 - 2

Low development cost

Alt. 1 - 2
Alt. 2 - 2

SELECTED APPROACH

Alternate 2 - nickel-cadmium battery, based on probability of no failure and low development cost.

Figure 1.2.1-5: BATTERY TYPE TRADE STUDY

TRADE STUDY SUMMARY SHEET | SOURCE OF REQUIREMENT | TRADE STUDY NUMBER & TITLE: VOLTAGE REGULATION

MATRIX OF DESIGN APPROACH | SELECTION

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

FUNCTIONAL REQUIREMENTS

The voltage regulator must convert variable voltage DC from the solar array to regulated voltage DC for the 2.4 KHz inverters.

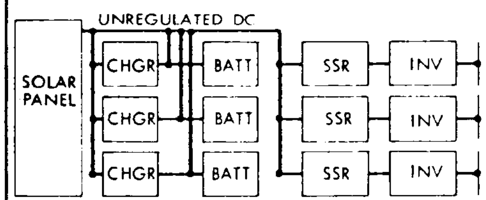
DESIGN REQUIREMENTS

1. Voltage regulation must be accomplished over the following range of solar array operating conditions of solar intensity and array temperature:
   a. 140 mw/cm^2 @ 10 to 80°C.
   b. 50 mw/cm^2 @ 16 to -170°C.
2. No single failure will result in loss of spacecraft power.
3. Cooperative multichannel design is preferred to alternate path or block redundancy.
4. The voltage regulation system should make optimum use of the maximum power capability of the solar array under near Mars conditions (i.e. 50 mw/cm^2 @ -10°C.
5. The system should provide for an array voltage high enough to charge the Voyager batteries to 57 volts through a bucking-type battery charger.

COMPETING CHARACTERISTICS

1. Simplicity
2. Performance
3. Low development risk (proven concept)
4. Simple thermal control
5. Low weight

ALTERNATE 1 — SERIES SWITCHING REGULATORS (SSR)



DESCRIPTION

The solar array and battery provide power at voltages ranging from 37 volts (battery minimum discharge voltage) to 100 volts (array transient voltage following occultation at Mars). The series switching regulators step down the raw DC power to obtain 35 volt DC for the inverters.

DISCUSSION

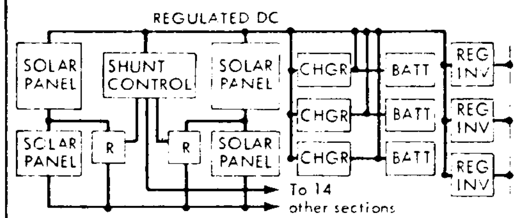
PRO:

1. No transient disturbances occur during transfer between battery power and solar power.
2. A simple subsystem design results.
3. The three regulators are isolated from each other with respect to voltage control.
4. Small variation in thermal dissipation throughout mission.

CON:

1. Voltage on the unregulated DC bus will vary from 37 to 100 volts.
2. Less efficient utilization of the solar array.

ALTERNATE 2 — PROPORTIONAL CASCADE SHUNT REGULATORS (PCSR)



DESCRIPTION

The solar array is divided into 16 parallel sections each having its own shunt regulator (R). The shunt regulator on each section controls section voltage by proportional power transistors shunting 35% of the panel section. The proportional power transistors are activated sequentially by a central control that senses the DC bus voltage. Due to the difference in shunt regulated voltage (62 volts) and nominal battery voltage (45.6 volts), regulated inverters are required.

DISCUSSION

PRO:

1. All DC power is inherently regulated.
2. About 4% less solar panel area may be required based on the following assumptions:
   a. The shunt regulator control and leakage loss is about 1%.
   b. The SSR is adjusted to operate 5% off the array maximum power point for stability.
   c. The relative efficiency advantage of a regulated inverter over an SSR and inverter can be gained in either system.

CON:

1. Approximately 11 pounds heavier.
2. More complex.
3. Requires regulated inverters because of difference between battery and shunt regulated voltage.
4. Large variation in thermal dissipation from near Earth to near Mars.

SELECTION

SIMPLICITY
Alt. 1 - 1
Alt. 2 - 2

PERFORMANCE
Alt. 1 - 1
Alt. 2 - 1

PROVEN CONCEPT
Alt. 1 - 1
Alt. 2 - 2

THERMAL CONTROL
Alt. 1 - 1
Alt. 2 - 2

LOW WEIGHT
Alt. 1 - 1
Alt. 2 - 2

SELECTED APPROACH

Alternate 1 -

Series switching regulator based on simplicity and proven concept.

Figure 1.2.1-6: VOLTAGE REGULATION TRADE STUDY

TRADE STUDY SUMMARY SHEET

SOURCE OF REQUIREMENT

TRADE STUDY NUMBER & TITLE AC OR DC DISTRIBUTION

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

FUNCTIONAL REQUIREMENTS

Distribute power to spacecraft loads at the voltage and voltage regulation required by each of the load circuits or power supplies.

DESIGN REQUIREMENT

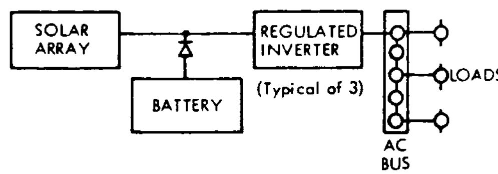
A single failure within the power conditioning equipment shall not cause loss of power to any one load.

COMPETING CHARACTERISTICS

1. Reliability
2. Subsystem Interface Simplicity
3. Low Development Cost
4. Low Weight
5. Low Subsystem Losses

MATRIX OF DESIGN APPROACH

ALTERNATE 1 AC DISTRIBUTION



DESCRIPTION

Power to electronic loads is distributed from three regulated inverters, at 50 Vrms, 2400 Hz, square-wave. Conversion to required dc voltages accomplished by transformer-rectifier units that are within the power utilizing equipment. Two of the three inverters can carry the loads.

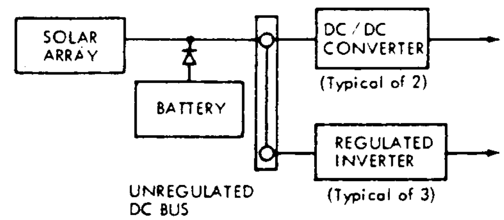
PRO

1. Power distribution is accomplished through a relatively simple subsystem interface.
2. Only an inverter, having output characteristics similar to the Mariner 1969 inverter, need be developed.
3. Inverter rating is relatively insensitive to minor power demand changes.

CON

1. Power subsystem losses possibly increased by 10 to 20 watts. Losses represent 1 to 2% of solar array capability.
2. EMI radiated by 2400 Hz square-wave current in power wires may affect sensitive circuits.

ALTERNATE 2 PARTIAL DC DISTRIBUTION



DESCRIPTION

48 watts at plus 4.5 Vdc is supplied to the computing and sequencing subsystem by two dc/dc converters. Other electronic loads are supplied 50 Vrms, 2400 Hz, square-wave power from three inverters. (DC power at over 20 positive and negative voltages is required for these other loads, each at a power level of less than 10 watts. Direct supply of each of these loads is not competitive with central ac distribution.)

PRO

1. Size of computing and sequencing power supply reduced.
2. Possible 10 to 20 watt reduction in power subsystem losses.

CON

1. DC/DC converter needs to be developed and qualified.
2. More complicated subsystem interface, particularly in protection, grounding, and load changes.
3. DC/DC converter ratings would have to be changed whenever loads increase during spacecraft development.

SELECTION

Reliability
Alt. 1 - 1
Alt. 2 - 1

Simplicity
Alt. 1 - 1
Alt. 2 - 2

Cost
Alt. 1 - 1
Alt. 2 - 2

Weight
Alt. 1 - 1
Alt. 2 - 1

Losses
Alt. 1 - 2
Alt. 2 - 1

SELECTED APPROACH

Alt. 1 - AC Distribution - on the basis of simplicity and cost.

Figure 1.2.1-7: AC OR DC DISTRIBUTION TRADE STUDY

TRADE STUDY SUMMARY SHEET

SOURCE OF REQUIREMENT

TRADE STUDY NUMBER & TITLE: DIODE FAULT ISOLATION OR FAIL-SAFE SOLAR ARRAY DESIGN

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

FUNCTIONAL REQUIREMENTS

The solar array and batteries must supply continuous unregulated DC power to the DC bus during sun-illuminated and sun-occulted periods of the spacecraft mission.

DESIGN REQUIREMENTS

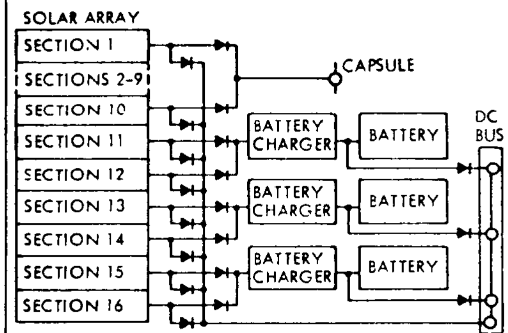
1. During periods when the spacecraft is occulted from the Sun, the capsule load must be isolated from the batteries in order to maintain the battery loads at an acceptable level.

COMPETING CHARACTERISTICS

1. Power dissipation
2. Subsystem reliability

MATRIX OF DESIGN APPROACH

ALTERNATE 1 — DIODE LOGIC FAULT ISOLATION



DESCRIPTION

The solar array is divided into 16 sections each isolated from the others by diodes.

DISCUSSION

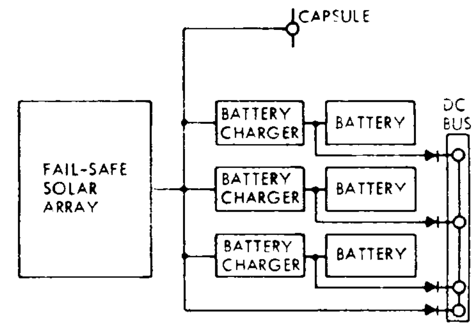
PRO:

1. Short-circuit failure of any section results in loss of no more than 1/16 of the array power capability.
2. Short-circuit failure of any charger or the capsule bus results in loss of only those sections required to supply these loads.

CON:

1. Series diodes dissipate about 1% of all power supplied from the array.

ALTERNATE 2 — FAIL-SAFE ARRAY DESIGN



DESCRIPTION

Solar panel performance in previous space flights indicates that a fail-safe design has been achieved. A diode or diodes are still required to prevent the batteries from supplying the capsule load.

DISCUSSION

PRO:

1. No series diode power dissipated in supplying the relatively large power requirement of the capsule and battery chargers.

CON:

1. No isolation provided for possible short-circuits of battery chargers or capsule bus.
2. Requires use of a larger, hence less reliable, diode or diodes connecting the solar array with the DC bus.
3. Any fault on the solar array wiring will disable the spacecraft.

SELECTION

POWER DISSIPATION
- Alt. 1 - 2
- Alt. 2 - 1

SUBSYSTEM RELIABILITY
- Alt. 1 - 1
- Alt. 2 - 2

SELECTED APPROACH

Alternate 1 - Diode fault isolation, based on isolation of faulted power equipment.

Figure 1.2.1-8: DIODE FAULT ISOLATION OR FAIL-SAFE SOLAR ARRAY DESIGN TRADE STUDY

2) Identify the fully charged condition with a third electrode (anhydrode). Fully satisfactory charge control has not yet been obtained with this feature.

3) Measure ampere-hours discharged with a coulometer; then restore withdrawn ampere-hours plus a loss. Coulometers have been developed by Boeing, Gulton Industries, and Applied Physics Laboratory, but none have space experience at this time.

Early start of battery testing is important because it takes 18 months to confirm that a given battery design has the life required for the Voyager mission. If the first battery design fails in life test, then an additional period of up to 18 months is required to test a redesigned battery. Past attempts to accurately predict battery life on the basis of testing for a fraction of the required battery life have not been successful.

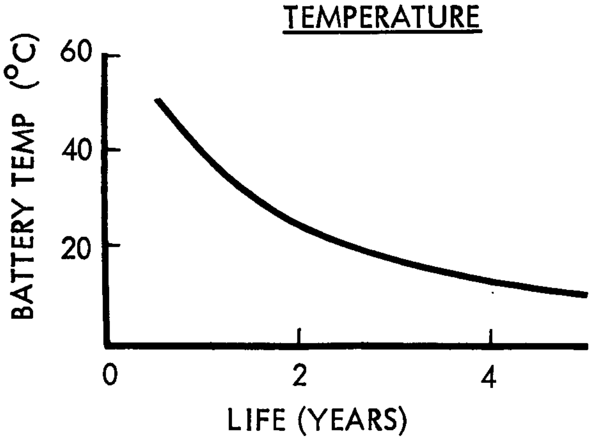
For these reasons, it is important to start battery life testing early. The development factors are summarized in Figure 1.2.1-9.

PROBLEM: Required Reliability 0.991 Power Subsystem
0.967 For One Battery (to Have a Reliability of 0.9965 for Battery Power - 2 of 3 Batteries required.)

Long Time (18 Months for Life Test)

BATTERY LIFE IS AFFECTED BY:

TEMPERATURE



CHARGE CONTROL

Constant - Current + Trickle
(Produces Overcharge, Heat)

Anhydrode or Third Electrode
(Not Yet Space-Proven)

Coulometer
(Not Yet Proven)

PROBLEM SOLUTION

Procure 20 AH Sealed Nickel-Cadmium Cells

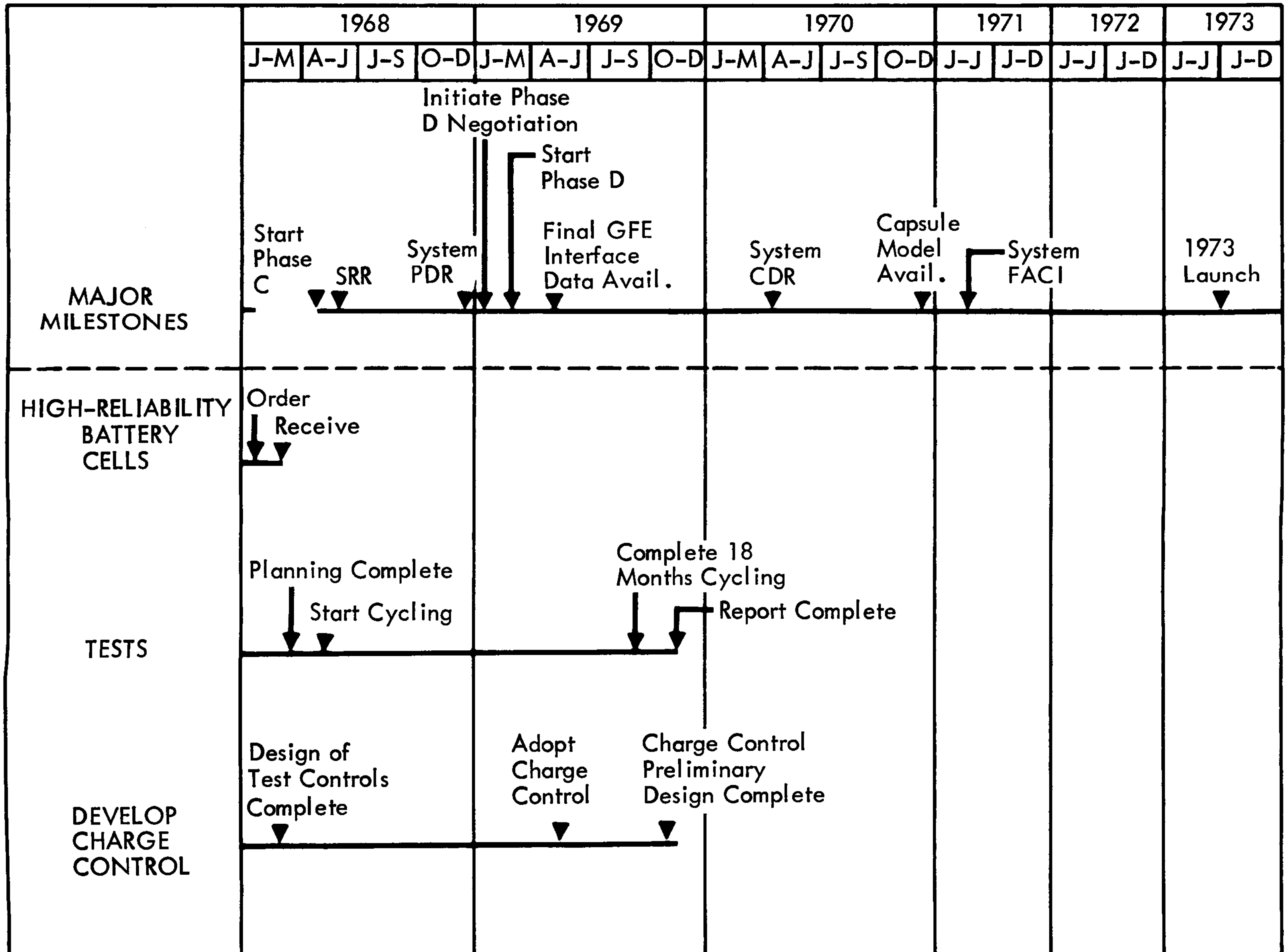
Conduct 18-Month Life Tests
Simulate Different Charge Techniques
Use Voyager Power Profile, Temperature

Figure 1.2.1-9: HIGH-RELIABILITY BATTERY DEVELOPMENT FACTORS

The development schedule is shown in Figure 1.2.1-10.

1.2.1.8 Growth Potential

Spacecraft electrical load growth can be accommodated (1) by using design reserves in the power subsystem, and (2) by providing more solar array area and higher-rated components.



	1968 J-M	1968 A-J	1968 J-S	1968 O-D	1969 J-M	1969 A-J	1969 J-S	1969 O-D	1970 J-M	1970 A-J	1970 J-S	1970 O-D	1971 J-J	1971 J-D	1972 J-J	1972 J-D	1973 J-J	1973 J-D	Total
LABOR & MAT'L ($)	32K	3K	3K	3K	3K	6K	6K	6K	3K										65K
CAP. FACILITIES ($)	0	0	0	0	0	0	0	0	0										0

Figure 1.2.1-10: HIGH-RELIABILITY BATTERY DEVELOPMENT SCHEDULE

The 290-square-foot solar array is capable of supplying 15% more power than demanded during the battery recharge period following spacecraft orbit trim (termed "S/C Orbit Recharge 1" in Figure 1.2.1-1). The solar array design margin during other flight regimes is larger than 15%. The regulated inverters are rated 10% above their individual maximum steady-state power demand. The spacecraft batteries undergo 52% depth of discharge in Mars orbital operation when two batteries are functioning. This is the deepest discharge that can reasonably provide full mission life for the batteries; hence it contains no design margin. However, with all three of the batteries functioning, the depth of discharge is only 34%, providing margin for load growth. The predicted mission reliability of 0.967 for one battery suggests that all three batteries will function the whole duration of 10 out of 11 flights.

Higher power capability can be provided by adding deployable solar panels to the spacecraft. For example, up to 330 square feet of solar array area (three deployed panels), providing approximately 990 watts of additional source power, can be added. By adding a fourth 20 ampere-hour battery and fourth battery charger, an additional 910 watt-hours of energy storage can be provided. The resulting changes to the power switching and logic unit and the power subsystem monitor are minor. Additional power capability at the a.c. busses can be provided by adding a fourth 230-watt inverter, or using three higher-output inverters. The choice will depend on the availability of suitable qualified power transistors, and modification costs.

BLANK

1.2.2 Guidance and Control Subsystem

1.2.2 Guidance and Control Subsystems

The Voyager spacecraft requires three-axis attitude stabilization to successfully accomplish the mission. The Sun-Canopus celestial references are augmented by an inertial reference system for maneuvers and periods of occultation. Control signals from these references are processed by the attitude control electronics and then command the nitrogen cold-gas thrusters which apply the control torques to the spacecraft. To control the spacecraft during the propulsive portion of a maneuver, the thrusting engine is gimbaled by electromechanical actuators that are commanded from the inertial reference unit and attitude control electronics. The following equipment is utilized to accomplish these functions:

1) The Voyager Canopus sensor will be an instrument similar to those used on Mariner IV, Mariner Venus 67, and Lunar Orbiter. These sensors use an image dissector tube, allowing a wide field of view (32 degrees in pitch by 4 degrees in roll) without requiring mechanical gimbaling. The Canopus sensor will be mounted in the guidance and control assembly near the outboard edge of the solar panel. Only the required portion of the optical field of view is scanned to generate error signals for spacecraft control. Since the Canopus cone angle variation with time is known, the desired sector can be selected through the computer and sequencer.

2) Photovoltaic Sun sensors are chosen as the preferred type. Essentially three Sun sensor outputs are provided. A coarse Sun sensor having a 4π steradian field of view is used for acquisition. An acquisition signal is generated whenever the Sun is within 3 degrees of the fine Sun sensor null. This signal is used to control the coarse-fine switching logic. A fine Sun sensor provides signals for pitch and yaw axis control about the sunline for normal limit cycle operation. The design selected does not compensate for the change in Sun sensor scale factor as the spacecraft travels from Earth (1 A.U.) to Mars (1.67 A.U.). However, the effect may be compensated by a selectable gain amplifier in the autopilot electronics and commanded through the computer and sequencer, if desired, as shown in the trade study summary, Section 1.2.2.6.

3) Two inertial reference units (IRU) will be provided, each having three orthogonally mounted gyros. Single degree of freedom, floated, rate-integrating gyros are chosen as the preferred instrument based on the availability and maturity of development for this type gyro for application to space vehicle attitude control. Each gyro in each IRU is individually switchable to allow per-axis redundancy rather than IRU redundancy. The heater power for each IRU is switched independent of the gyros to allow heater operation with the gyros off. The gyros in each IRU are capable of operating either in the rate or position mode. The rate mode is used for rate damping and maneuvers while the position mode is used for limit cycle operation in the attitude hold mode.

IRU channel malfunctions are detected by monitoring the following:

Gyro spin motor
Gyro temperature
Gyro loop output (rate and position)

Spin motor and temperature control failures are detected by monitoring spin motor current and gyro temperature sensor resistance. Gyro loop output can be compared to the celestial sensor error signals to determine malfunctions. Selection and switching are controlled by ground command or by on-board malfunction detection when provided.

4) An accelerometer is used to measure the spacecraft velocity change along the thrust axis during engine firing. The velocity increment is measured in the computer and sequencer where the accelerometer output is integrated. Thrust is terminated when the velocity change reaches the predetermined required value. A timed backup approach, based on the 3σ variations of burn-time tolerances, is used to obtain degraded performance in the event of accelerometer failure. In addition, a real-time backup redundant ignition command can be transmitted from the ground, timed so as to be received at the spacecraft during the normal burn time. This would be followed by a redundant terminate burn command timed for receipt at the spacecraft after the end of a normal burn.

5) Electromechanical actuators position the LMDE engine gimbal assemblies. The actuator is powered by a variable-speed d.c. motor driving a ball jackscrew through reduction gearing. Rate and position sensors provide feedback signals for closed-loop operation.

6) Two limb and terminator sensors are provided to detect the vehicle crossings of the Mars illuminated limb and the crossing of the Mars terminator. The sensor fields of view are directed 90 degrees from the sunline on either side of the spacecraft. The field of view is narrow and fan-shaped to cover the range of orbits desired.

7) Reaction Control -- The selected concept is two standby redundant cold-gas nitrogen reaction jet systems similar to the system on Lunar Orbiter. Nitrogen is stored in four tanks fabricated from Ti-6Al-4V titanium alloy. Each gas system has eight thrusters; pitch and yaw are 0.45 pound thrust, and roll thrusters are 0.20 pound thrust working in force couples. A schematic of the subsystem is shown in Figure 1.2.2-1. A maneuver rate of 0.2 deg/sec has been selected as a compromise between conserving propellant versus a capability for reasonably short maneuver times required by battery life considerations. The minimum angular acceleration of 0.01 deg/sec^2 in

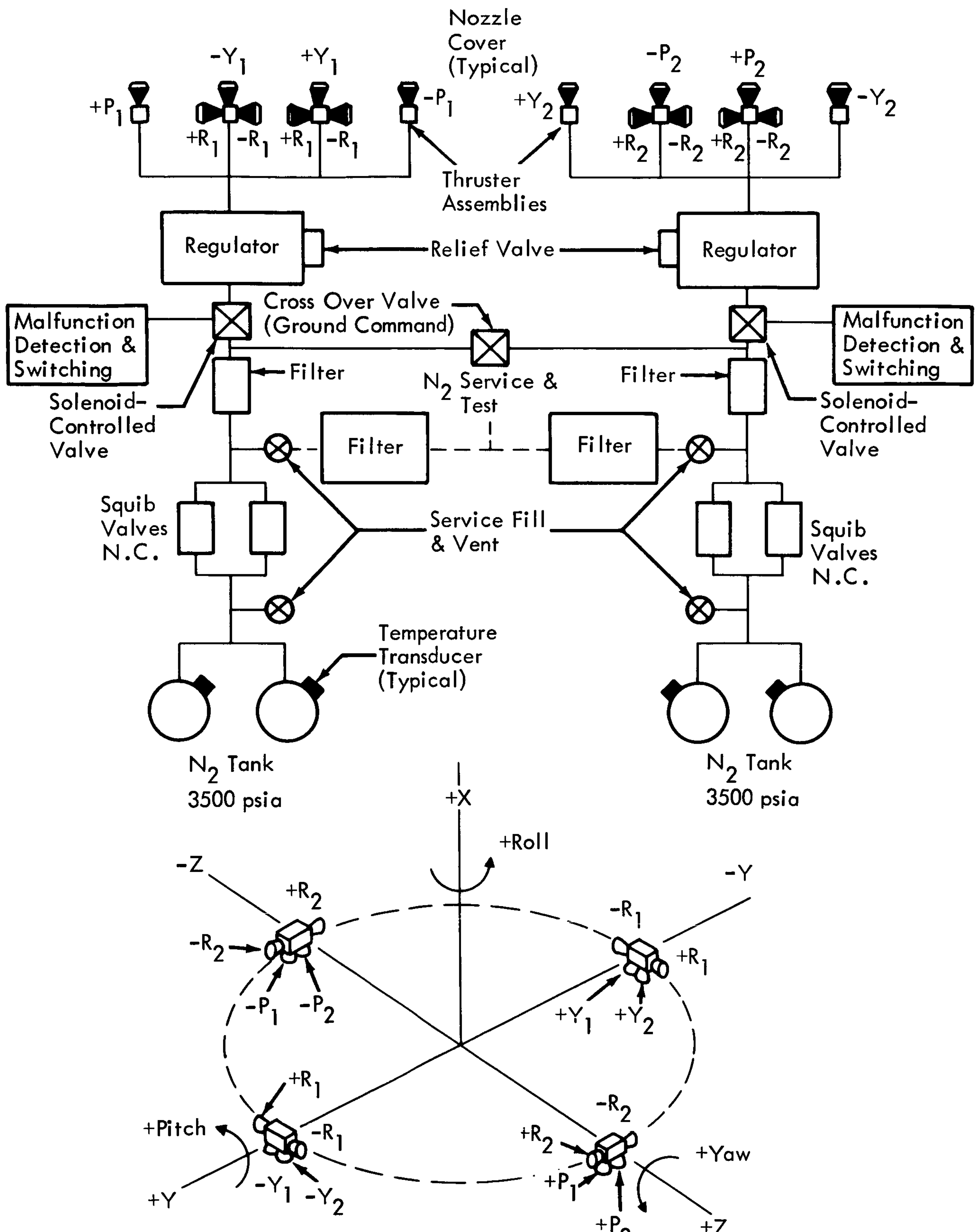


Figure 1.2.2-1: REACTION CONTROL SCHEMATIC

pitch and yaw is a compromise between limit cycle fuel consumption and avoiding excessive angular overshoot at end of a maneuver. An additional requirement in the roll axis is that the control torque must be sufficient to contain the transients that may occur due to misalignments at engine startup, and steady-state swirl torques. Nitrogen gas requirements, shown in Table 1.2.2-1, are based on a specific impulse of 68 seconds. This value is a conservative figure for -40°F gas. Tests indicate that thruster performance may be degraded as much as 30% for very short pulse operation. This effect is virtually eliminated by the minimum on-time (single-shot) device. Exact length of the single-shot time will be subject to refinement after valve characteristics, deadband width, and solar disturbance torques are better defined. Twenty milliseconds has been used here and in the autopilot design.

The nitrogen gas will be filtered and loaded aseptically to reduce microbial contamination.

8) Alignment -- All attitude reference sensors with the exception of the coarse sun sensors will be mounted in a common package, thus simplifying their relative alignment. The guidance and control package alignment relative to the spacecraft axes will be determined to within 0.04 degree. The guidance and control package installation will be within 0.15 degree of the spacecraft axes. Sensor axes will be aligned to the guidance and control package to within 0.01 degree. Reference mirrors on the IRU, fine Sun sensors, and Canopus sensors are used for sensor mutual alignment and guidance and control package alignment to the spacecraft. Measurement accuracy in optically aligning the sensors in the guidance and control package will be on the order of 0.1 minutes of arc and 0.01 degree in the mounting of the guidance and control package to the spacecraft. In addition to mechanical alignment, alignment of the attitude control electronics is also necessary. Primarily, this involves trimming out amplifier null offsets.

1.2.2.1 Design Constraints and Requirements

The design constraints and requirements are shown in Table 1.2.2-2.

The control concepts recommended are essentially the same as those resulting from the Task B studies. The principal differences are in the manner of mechanization: the most significant of these are listed below:

1) Single-axis gyros were selected instead of the two-axis free-rotor gyro because of the space-proven status of the former.

2) Thrust vector control is achieved by gimbaling the LMDE instead of by secondary liquid injection in the nozzle of the solid propellant engine because of the change to all liquid propulsion.

Table 1.2.2-1: NITROGEN ALLOCATION

TRANSIT	IMPULSE (lb-sec)	ORBIT	IMPULSE (lb-sec)	TOTALS	IMPULSE (lb-sec)
1. Initial Sun Acquisition 1^{o}/sec Separation Rates	137.3	1. Orbit Insertion Roll and Yaw Maneuver	117.3	Mission	2031.1
2. Initial Canopus Acquisition 1^{o}/sec Separation Rates	57.5	2. First Orbit Trim Roll and Yaw Maneuver	105.4	Required for Redundancy (Total x 2)	4062.2
3. First Midcourse Roll and Yaw Maneuver	114.6	3. Second Orbit Trim Roll and Yaw Maneuver	105.4	Residuals	136.0
				Tankage Requirement	**4198.2**
4. Second Midcourse Roll and Yaw Maneuver	132.1	4. Limit Cycle Derived Rate, $\pm 0.3^{o}$ for 180 days	340.0	Nitrogen Weight (I_{sp} = 68 sec)	62 lb
5. Third Midcourse Roll and Yaw Maneuver	132.1	5. Limit Cycle Gyro Rate, All Maneuvers	34.7		
6. Limit Cycle Derived Rate, $\pm 0.3^{o}$ for 225 Days	93.7	6. Sun Reacquisitions 137 Occultations	8.8		
7. Limit Cycle Gyro Rate, All Maneuvers	66.0	7. Canopus Reacquisition 308 Occultations	27.4		
8. Celestial Reacquisitions	1.3	8. Celestial Reacquisitions After Maneuver	2.8		
9. Disturbance Torque Solar Pressure	248.0	9. Disturbances	246.7		
10. Leakage	33.3	10. Leakage	26.7		
TRANSIT TOTAL	1015.9	ORBIT TOTAL	1015.2		

Table 1.2.2-2: DESIGN CONSTRAINTS AND REQUIREMENTS

CONSTRAINT/REQUIREMENT	VALUE
Reliability Allocations	0.983
Spacecraft Angular Rate (Following Separation of S/C and Booster)	≤1.0 deg/sec
Roll Rate During Telemetry Phase Lock Acquisition	≤0.2 deg/sec
Maneuvers	
● Accelerometer Range	±2.0 g
● ΔV Proportional Error (Orbit Insertion)	≤0.15% (3 σ)
● ΔV Resolution (Excluding Engine)	≤0.02m/sec (3 σ)
● Pointing Accuracy Allocation	≤2.4 deg (3 σ)
Limit Cycle	
● Deadband (Based on Allocation From Antenna Pointing Accuracy Requirement)	± 0.3 deg
● Rate (Based on Science Scan Platform Requirements)	<0.0074 deg/sec
Maximum Loss Time of Celestial References	
● Occultation: Sun	2.1 hr max
● Occultation: Canopus (see Figure 1.2.2-7)	5.9 hr max
● Maneuvers: Sun	2.1 hr max
● Maneuvers: Canopus	2.5 hr max
Gyro Stability (Drift)	±0.3°/hr (3 σ)
Attitude Reference Sensing Accuracy	
● Sun	±0.15 deg (3 σ)
● Canopus	± 0.3 deg (3 σ)
Limb-Terminator Sensing Accuracy	1.0 deg

3) Standby redundancy concepts are used with strong dependence on the operations personnel for command, diagnosis, and selection functions to improve the predicted reliability over the single-thread system.

4) The guidance scan platform and planet sensor used for approach guidance refinement were deleted as a result of guidance and navigation studies indicating that it did not enhance the operation.

5) The Earth sensor used previously to verify initial Canopus acquisition was deleted. Instead, the technique of identifying Canopus by interpretation of star map information on the ground is recommended with a backup roll orientation identification capability by means of the high or medium gain antenna.

1.2.2.2 Functional Description and Performance Characteristics

The guidance and control subsystem performance characteristics are summarized in Table 1.2.2-3.

Attitude Control Loop -- The function of the attitude control loop is to provide proper orientation and stabilization of the spacecraft during all unpowered phases of the mission. A block diagram of the attitude control loop is shown in Figures 1.2.2-2 and 1.2.2-3.

Attitude sensors provide electrical signals that are proportional to the spacecraft attitude deviation from the reference orientation. These error signals are processed by the attitude control electronics into jet solenoid firing pulses. The control torques from the cold-gas thrusters cause vehicle rotations which are detected by the reference sensors, thus closing the loop.

An attitude rate signal is required for stable operation of the attitude loop. A rate signal is not always available from the gyros because they are turned off during cruise and are operating in the position (rate-integrate) mode during inertial-hold periods. In the proposed design, two types of derived rate compensation are used. Lag network feedback around the switching amplifier ensures efficient limit-cycle operation for attitude hold. Lead-lag compensation, when applied to the gyro position signal, results in a signal that is proportional to position plus differentiated position (rate). This signal provides good performance when the spacecraft is converging from attitude-maneuver to attitude-hold operation. For simplicity and reliability both derived-rate circuits remain connected at all times. The effect of lead-lag compensation on limit-cycle operation is negligible. The effect of lag around the switching amplifier, while not negligible, is acceptable with the proper selection of parameters and preferred over the alternative of additional switching.

A "one-shot" device follows the switching amplifier. This ensures that the jets remain on for a minimum time duration (nominally 20 milliseconds) to produce a reliable, efficient, and predictable minimum impulse bit.

Table 1.2.2-3: GUIDANCE AND CONTROL SUBSYSTEM PERFORMANCE CHARACTERISTICS

SYSTEM PERFORMANCE	VALUES
Thrust Vector Error (3σ)	
Including 3σ Attitude Maneuver Error of	1.22 deg
Midcourse Maneuver	1.86 deg
Orbit Insertion	1.86 deg
Orbit Trim	1.86 deg
Gain Margins	> 6 db
Attitude Control	
Limit Cycle Dead Band	± 0.3 deg
Maneuver Rate	∓ 0.2 deg/sec
Limit Cycle Rate	≤ .0005 deg/sec

SYSTEM PARAMETERS	
Attitude Control (Rate Gain/Position Gain)	10/1 sec
TVC (Double Lead-Lag)	
Lead Time Constants	0.33, 0.2 sec
Lead to Lag Ratio	10/1
Autopilot Gain	4 (nominal) 1 (Alternate)
Pseudo-Rate Network	
$\frac{K_D E_A}{K_S}$	0.6 deg
Time Constant	
Charge	8 sec
Discharge	16 sec

IRU	
Rate Mode	
Linear Rate Capability	3 deg/sec
Accuracy	.04% at 0.2 deg/sec
Frequency Response	10 Hz
Position Mode	
Input Angle Range	± 5 deg
Acceleration Insensitive Drift	0.3 deg/hr max
Acceleration Sensitive Drift	2 deg/hr/g max
Frequency Response	26 Hz

ACCELEROMETER	
Range	±2.0 g
Bias	2×10^{-4} g
Scale Factor / Linearity Error	0.1%
Pulse Scaling	0.04 M/sec/pulse

CANOPUS SENSOR	VALUES
Roll Field of View	4 deg
Pitch Field of View	5 overlapping 11 deg steps, 32 deg total
Signal to Noise Ratio	20:1 minimum
Null Accuracy	0.02 deg
Stray Light Attenuation	Capable of tracking Canopus to within 15 deg of illuminated limb of Mars

SUN SENSOR	
Fine Sensor	
Null Accuracy	0.02 deg at 1.0 AU
Linear Range	± 3 deg
Time Constant	< 5 ms
Acquisition Signal	100 mv at 3 deg (1.0 AU)
Coarse Sensor	
Field of View	Spherical
Null Error	< 2 deg

LIMB TERMINATOR SENSOR	
Field of View (Total)	1.5 x 92 deg, each unit
Accuracy	± 1 deg

REACTION CONTROL		
Thrust Level		
Pitch and Yaw	0.45 lb	
Roll	0.20 lb	
Total Impulse (Basic Mission)	2031 lb-sec	
2 x redundancy allowance	4062 lb-sec	
Specific Impulse	68 seconds	
Leakage Total	40.0 scc/hr	
Angular Acceleration	Transit	Orbit
Pitch	0.01 deg/s^2	0.047 deg/s^2
Yaw	0.01 deg/s^2	0.052 deg/s^2
Roll	0.01 deg/s^2	0.027 deg/s^2

THRUST VECTOR CONTROL ACTUATOR (LMDE)	
Force	250 lb
Velocity	3.8 in./second
Stroke	3.3 in
Frequency Response	5 Hz

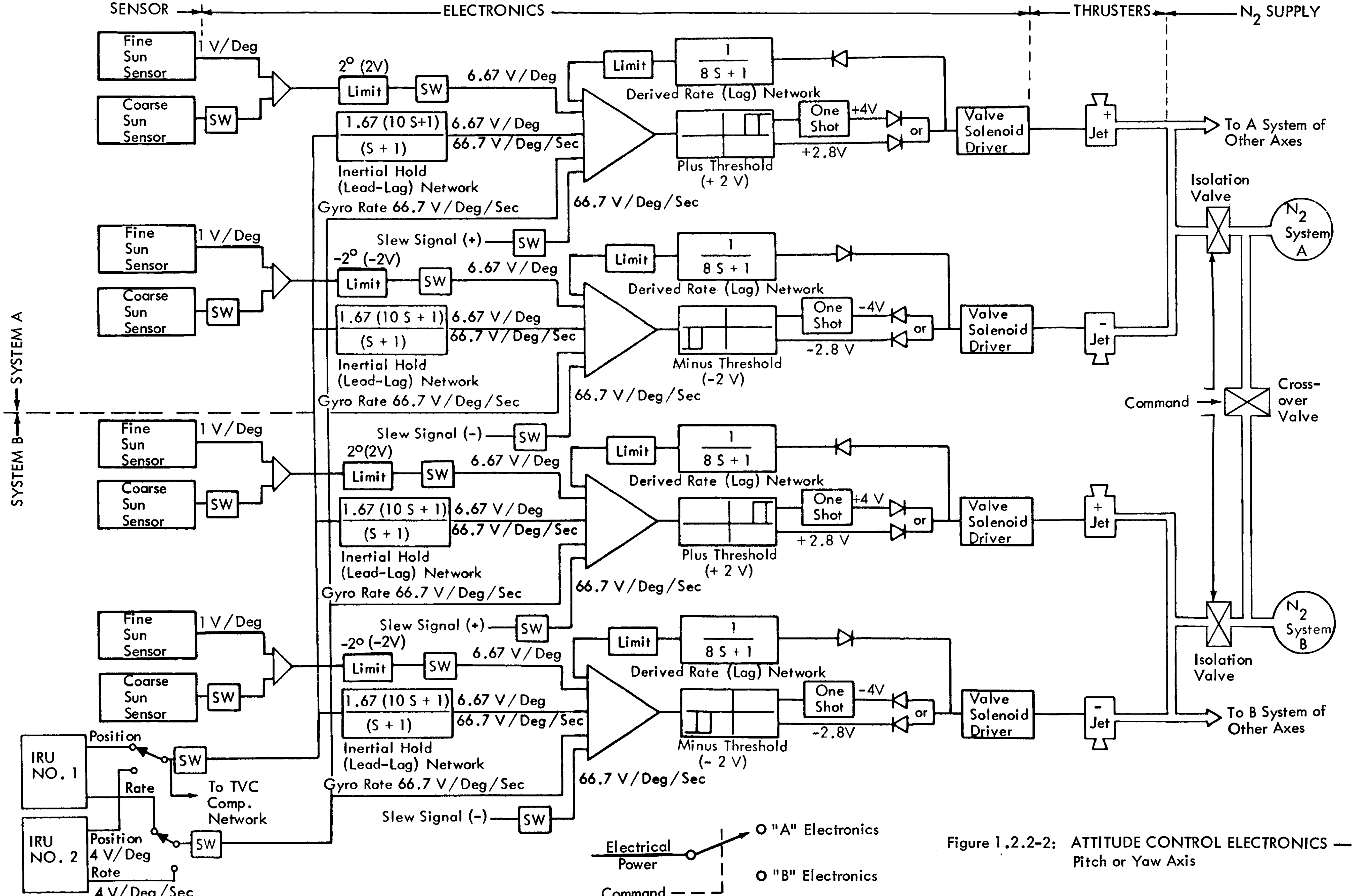


Figure 1.2.2-2: ATTITUDE CONTROL ELECTRONICS — Pitch or Yaw Axis

FOLDOUT FRAME 1

FOLDOUT FRAME 2

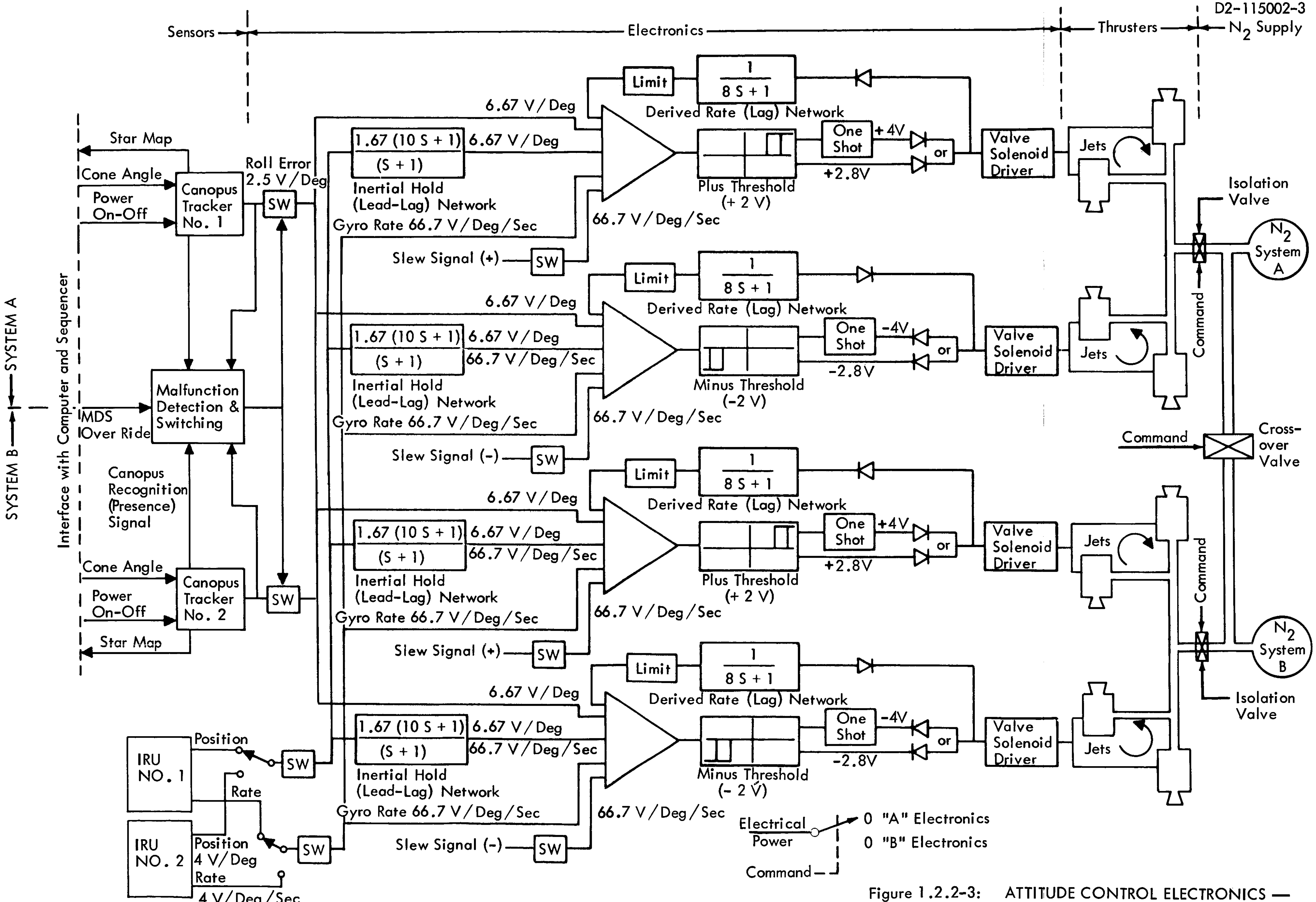


Figure 1.2.2-3: ATTITUDE CONTROL ELECTRONICS — Roll Axis

Parallel control paths are provided from the Sun sensor to the valve solenoid for both plus and minus halves of each system. This provides some degree of fail-passive operation. In the event of a partial-on failure in one side, the error signal generated prevents spinning up the spacecraft. The standby system could then be activated by ground command and the flight could proceed in a normal manner. Precise alignment must be performed between the plus and minus side of each control channel, including sensors and electronics.

Thrust Vector Control Loop -- The basic design constraints and requirements governing the thrust vector control are presented in Table 1.2.2-2. Of the pointing accuracy requirements of 2.4 degrees, for the nominal mission, 1.34 degrees are allocated to attitude maneuvering and 2 degrees for thrust vector control (TVC) loop errors. The major parameters governing the TVC loop error are engine gimbal to center-of-mass distance (L), center-of-mass offset, and commanded engine angle to vehicle angle gain. The relationship of the first two parameters for unity gain are presented in Figure 1.2.2-4. The band of center-of-mass offset effects is bounded by 0.5 inch and 1 inch offsets for the lower and upper curves, respectively. Allowances for such considerations as science scan platform or other appendage deployment may require revisions when a final design is evolved.

An analytical breakdown of the attitude maneuver and thrust vector pointing errors is presented in Section 1.1.5.

Stable, accurate performance is required in the presence of vehicle structural flexing and propellant sloshing. From these considerations the thrust vector control loop design imposes requirements on the vehicle configuration. These requirements are shown in Figure 1.2.2-5. The vehicle engine configuration, in turn, imposes the following sizing requirements on the thrust vector actuators in order to achieve adequate response:

Parameter	Value (LMDE)
Gimbal friction	650 in. lb
Propellant hose compliance	130 in. lb/deg
Thrust misalignment	0.125 in.
Gimbal acceleration	± 200 deg/sec^2
Gimbal rate	± 12.5 deg/sec
Gimbal deflection	± 5.5 deg

These requirements are met by the thrust vector control loop shown in Figure 1.2.2-6. Vehicle attitude information is provided by the gyros of the inertial reference unit operating in rate-integrate mode. Loop stabilization is provided by double lead-lag compensation and a gain of 4 degrees δc per degree θ.

Gyro output high-frequency noise is filtered by a double-lag network. The gain of 4 is utilized for all normal modes of operation. An additional gain of 1 is provided for the emergency condition of capsule-off/orbit-insertion. The actuator

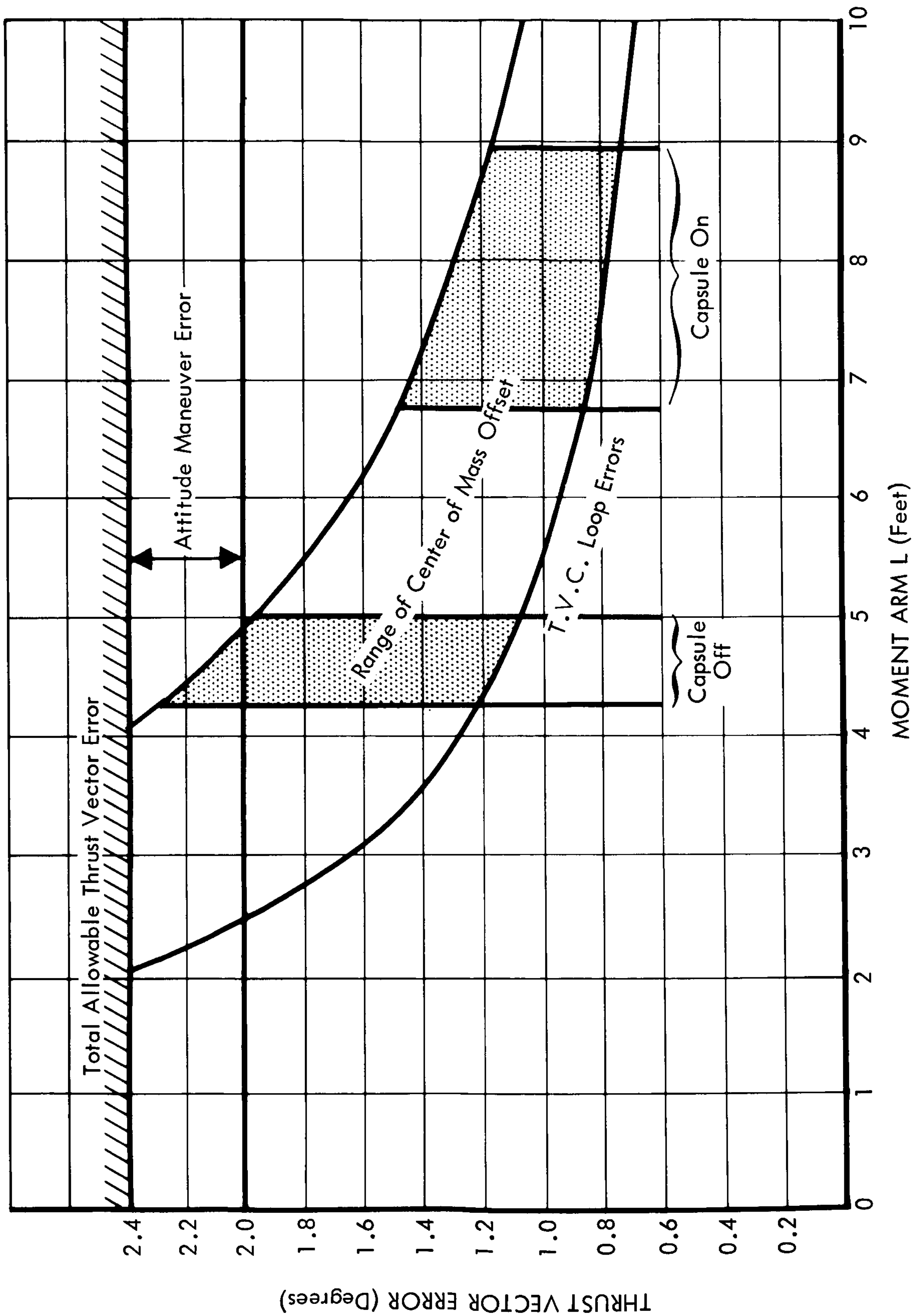


Figure 1.2.2-4: THRUST VECTOR CONTROL ERROR REQUIREMENTS

- Gimbal Lever Arm ($L > 5$ ft Capsule On)
- Slosh Damping ($\omega < 5$ R/S; ζ 0.01 to 0.2)*
- Center of Mass to Center of Tank ($P > O$) for Stable Slash Mode
- Center of Mass Offset (ΔZ C.M. < 1 inch)
- Rigidity ($\omega > 10$ R/S $\zeta > 0.01$)

*Note:
$\zeta = 0.2$ required for unstable slosh configuration. (Damping factor)

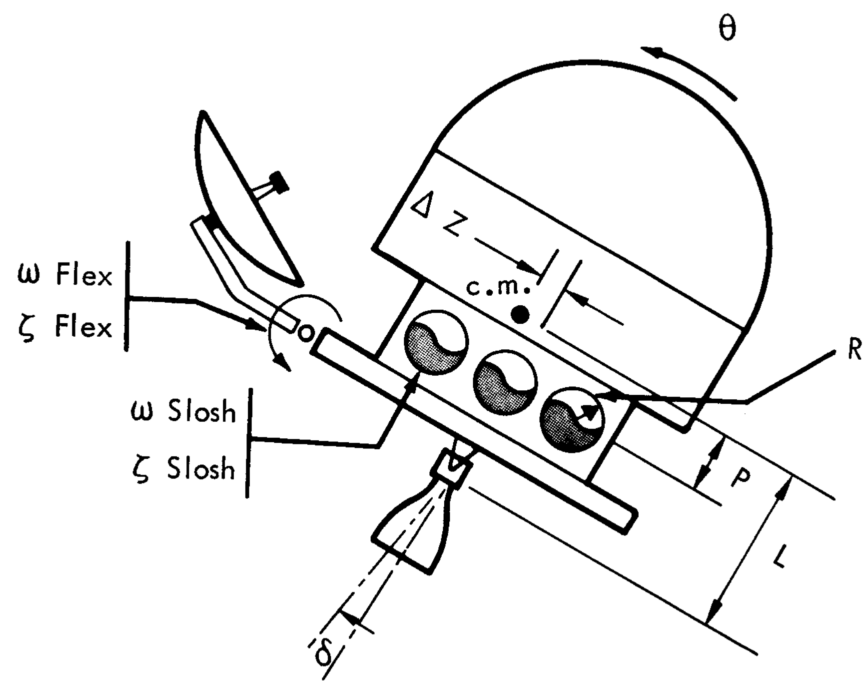


Figure 1.2.2-5: CONFIGURATION REQUIREMENTS IMPOSED BY THRUST VECTOR CONTROL LOOP DESIGN

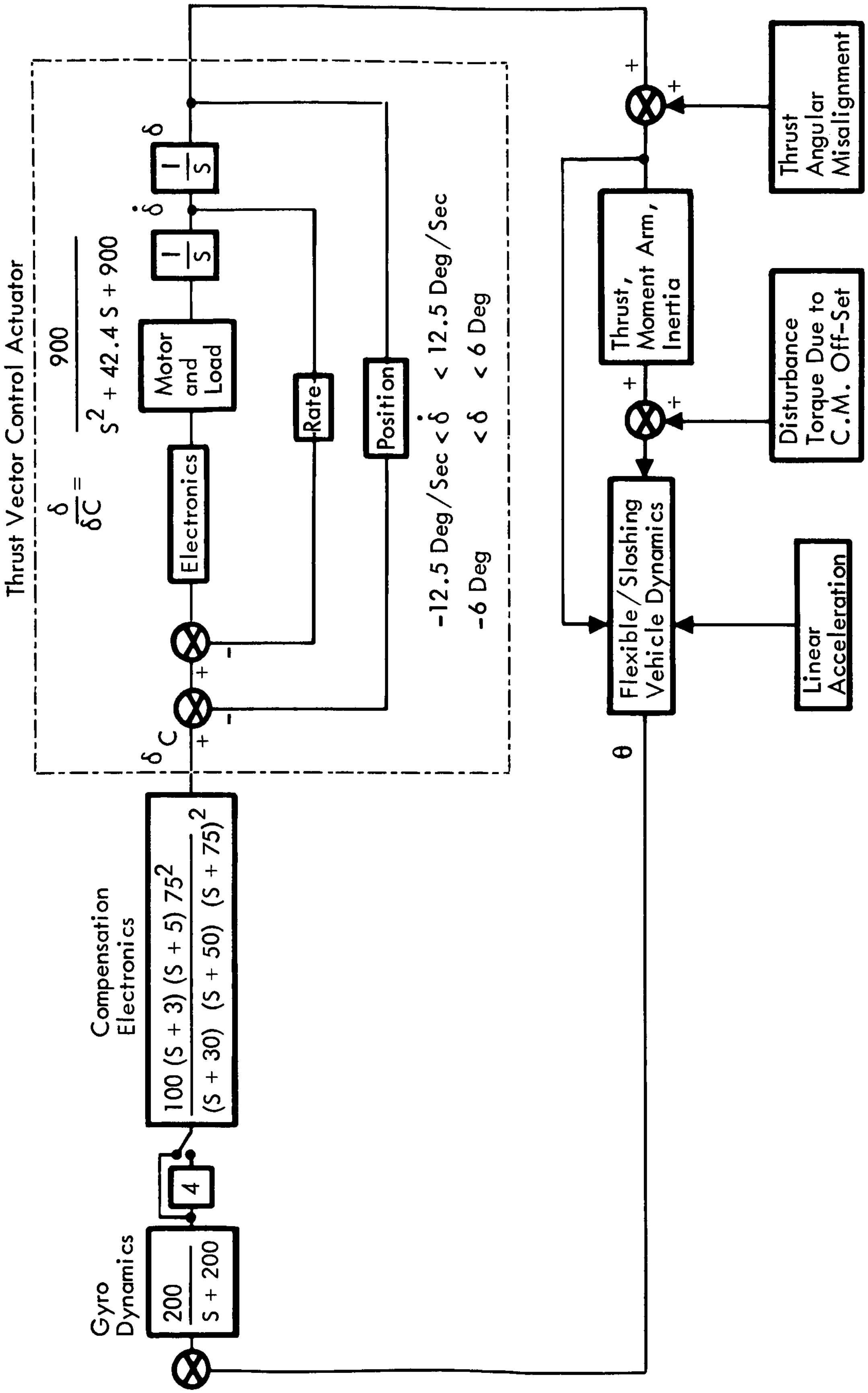


Figure 1.2.2-6: THRUST VECTOR CONTROL SYSTEM-SINGLE THREAD

consists of a variable-speed d.c. motor with gear reduction driving a ball jack-screw. Rate and position feedback signals are utilized to provide the desired response.

The roll reaction control system provides 48 in.-lb of control torque. All LMDE startups are made at low thrust. Assuming both pitch and yaw gimbals were hardover at engine ignition, this would induce a transient roll rate increment of 0.0313 deg/sec while the actuators are running to center. If the roll system were at the edge of its limit cycle, the resulting roll angle excursion would be an additional 0.05 degree. The maximum specified steady-state roll torque due to misalignments from the LMDE is 25 in-lb; however, the 3σ value is expected to be less than 15 in.-lb. Consequently, the normal attitude control roll system is adequately sized to provide satisfactory performance during powered flight.

Functionally, the thrust vector control loop provides vehicle attitude control during powered flight with the performance characteristics listed in Table 1.2.2-3. The LMDE is operated at 1050 pounds thrust for midcourse and orbit trim maneuvers. The LMDE is started at 1050 pounds, but is run at 9850 pounds for orbit insertion.

Mission Phases -- The guidance and control subsystem provides closed-loop control of the spacecraft during all mission phases. The paragraphs which follow give a brief description of how this is accomplished for each mission phase.

1) Sun Acquisition -- Separation rates are damped using gyro rate information. Coarse Sun sensors provide 4π steradian coverage for acquisition from any initial orientation. Pitch and yaw axes are slewed simultaneously, using Sun sensor position and gyro rate information until the reference orientation is attained. The Sun sensor output is limited to prevent excess slew rates for large attitude errors. The acquisition rate is between 0.17 deg/sec and 0.23 deg/sec. Control torques are provided by the cold-gas reaction jets. Roll rate is reduced to 0.03 deg/sec during Sun acquisition and the roll channel remains in the rate damping mode until the Canopus acquisition sequence.

2) Canopus Acquisition -- Acquisition of Canopus can be accomplished using two methods. When the spacecraft is near Earth, Canopus is acquired by rolling the vehicle about the sunline and telemetering the Canopus sensor intensity signal. The position of Canopus is identified by comparing the telemetered star trace with a previously prepared map and then commanding a vehicle roll maneuver to Canopus. In the second method, a roll search is commanded and the Canopus sensor is allowed to lock automatically on the first target falling within the bounds of preset intensity gates. These gates must necessarily be wide to allow for sensor gain changes due to photocathode non-uniformity and photomultiplier gain degradation. In the event the wrong target is selected, a ground override command is used to continue the search. Both the medium and high gain antennas provide corroborating information on spacecraft orientation. Because of the requirement to track Canopus while orbiting Mars, the Canopus sensor stray light baffling must be adequate to attenuate stray light reflected from Mars. In addition, a bright object sensor is required to prevent the possibility of damaging the photomultiplier tube from excessive light. The mounting position of the guidance and control package allows the use of a long baffle as well as eliminating glint from

the spacecraft itself. Significant Canopus sensor performance parameters are given in Table 1.2.2-3.

3) Cruise -- In this mode the gyros are turned off. The spacecraft is maintained in limit cycle operation about the celestial reference orientation. Position information is from the Sun and Canopus sensors. Damping is achieved using the pseudorate signal from the lag network feedback around the switching amplifier.

4) Attitude Maneuvers -- The maneuver method selected is basically the same as that used on Lunar Orbiter. During a maneuver, the gyro is in the rate mode. The rate output of the gyro is summed with the rate command signal in the autopilot. The gyro rate output signal is also digitized with a voltage-to-frequency converter in the computer and sequencer. An accumulator in the computer and sequencer counts the pulses and compares the total count with the commanded angle stored in the computer and sequencer memory. When coincidence occurs, the rate command signal is removed and the gyro is switched to the position mode. Convergence from the maneuver rate is then achieved using gyro position and lead-lag type compensation. This method was used on Lunar Orbiter with very successful results. The mean maneuver accuracy was about 0.1% of the maneuver angle. An important advantage of this maneuver method is that it provides on-board closed-loop control. It also provides for confirmation of execution of the maneuver by telemetering the contents of the computer and sequencer angle accumulator and the rate output of the gyro.

5) Midcourse Thrust Vector Control -- The LMDE will be centered sometime before the first midcourse thrusting period. This is achieved by having the gyros in the rate mode so that their position outputs, and consequently the input to the TVC actuator loops, are zero. Under these conditions power will be applied to the actuator causing the engine to center.

At the first midcourse engine firing, the LMDE will be operated at low thrust and there will be a startup transient due to engine thrust misalignment and vehicle center-of-mass offsets. If the final actuator design selection has an irreversible drive feature so that the engine will not move, all subsequent engine ignition transients will be small. Roll control during thrusting flight will be provided by the cold gas attitude control system operating in inertial hold mode.

Structural coupling and propellant slosh effects are not expected to be limiting factors on the midcourse TVC loop design.

6) Orbit Insertion Thrust Vector Control -- For orbit insertion, the normal LMDE ignition sequence calls for about 15 seconds of low thrust operation before switching to high thrust. Any TVC ignition transients will have reached their steady-state value by the time the engine is switched to high thrust.

Structural flexing and propellant sloshing effects will be at their maximum during the high thrust operation. However, structural coupling effects are not expected to cause limiting design constraints on the TVC loop, and the spacecraft propellant tankage is arranged to preclude naturally unstable slosh dynamics configurations during normal operation. Additional propellant slosh damping will be provided by internal baffling of the tanks so that satisfactory performance can be obtained should orbit insertion be required in a capsule-off configuration. Roll control is provided by the normal attitude control system.

7) Orbital Operations -- During orbital operations, the guidance and control subsystem provides spacecraft attitude orientation and stabilization in accordance with the mission requirements as described below.

During capsule separation, the guidance and control subsystem will provide a stable spacecraft orientation in the celestial reference attitude-hold mode. This will be the nominal operating mode during the orbital operations phase except for: periods of celestial reference occultation, attitude maneuvers as required by scientific experiments, and attitude maneuvers required for orbit trim firings of the LMDE. During these periods the guidance and control subsystem will operate in the attitude maneuver (slew) mode and the inertial hold mode. Gyros are kept running continuously after orbit insertion to avoid frequent start-stop cycles and to allow automatic inertial hold switching in the event of unpredicted celestial occultations or spacecraft upsets. Orbit trim TVC is provided in the same manner as for midcourse corrections since the LMDE is operated in low thrust mode only.

For the design orbit there are no occultations of Canopus. However, the limb of the planet does come close enough (within 8 degrees once each orbit) to the line of sight to Canopus to interfere with proper operation of the star tracker. The Mars-Canopus interference times are shown in Figure 1.2.2-7. The interference conditions (based on typical baffle design) is assumed to exist whenever the limb of Mars is within ±30 degrees of the spacecraft-Canopus line in the pitch direction and within ±20 degrees in the roll direction.

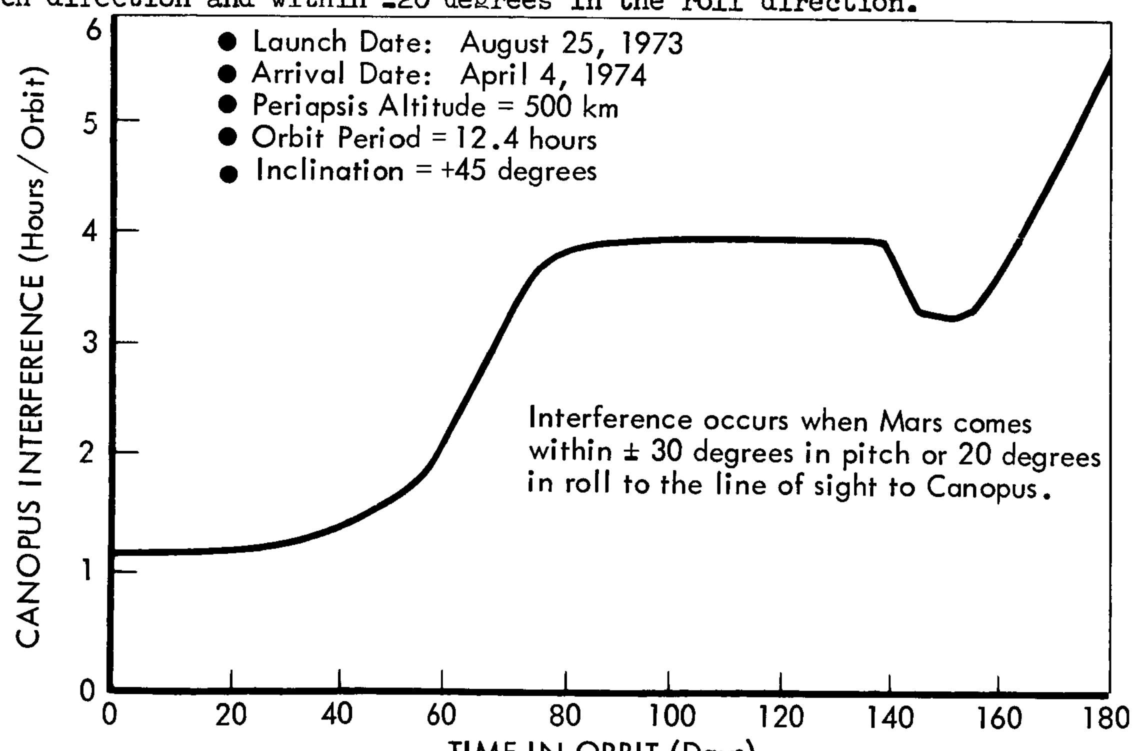


Figure 1.2.2-7: MARS CANOPUS INTERFENCE — Design Orbit

1.2.2.3 Physical Characteristics

The physical characteristics of the guidance and control subsystem equipment are shown in Table 1.2.2-4.

1.2.2.4 Interface Definition

Principal interfaces of the guidance and control subsystem are identified in Table 1.2.2-5. Physical interfaces with the spacecraft can be seen in Figure 1.2.2-8.

1.2.2.5 Reliability

The assessed reliability for guidance and control is 0.957. This is lower than the goal of 0.983 because of subsystem changes made as a result of trade study comparisons. These changes are:

1) Change to single degree of freedom, floated, rate-integrating gyros in lieu of the free rotor (two-axis) gyros previously used.

2) Change to a dual standby redundant concept for attitude control in lieu of the cooperative multichannel concept previously used.

3) Change to a gimbaled engine for thrust vector control in lieu of the secondary injection system and jet vanes previously used.

These changes are justified on the basis of weight, cost, availability, and performance considerations even though the assessed reliability does not meet the goal. Reliability assessments for the subsystem elements are shown in Table 1.2.2-6.

Table 1.2.2-6: GUIDANCE AND CONTROL RELIABILITY ASSESSMENT

SUBSYSTEM ELEMENTS	RELIABILITY ASSESSMENT	REL. GOAL
Inertial Reference Unit	0.9699	
Canopus Tracker	0.9981	
Thrust Vector Control	0.9985	
Attitude Control [1]	0.9898	
Accelerometer [2]	0.9944	
Limb & Terminator Sensors [2]	0.9973	
Total Subsystem	0.957	0.983

[1] Attitude control hardware includes the Sun sensors, autopilot electronics, and the reaction control subsystems.

[2] These items are not critical to mission success and are not included in total subsystem reliability assessment. The function performed by the accelerometer is critical, but it is backed up by a timer in the C&S subsystem and by real time commands from Earth.

Attitude Control Reliability -- The following attitude control redundancy concepts were examined for reliability:

1) Single thread

2) Dual standby

Table 1.2.2-4: GUIDANCE AND CONTROL SUBSYSTEM WEIGHT, SIZE, AND POWER

SUBSYSTEM ELEMENT	WEIGHT (Pounds)	DIMENSIONS (Inches)	POWER INPUT (Watts) AVERAGE	POWER INPUT (Watts) PEAK
G&C Equipment Bay Assembly	99.5			
Inertial Reference Unit	28	2 Units, 5.5x7x11	10 gyros 20 heaters	30* 40*
Canopus Sensors	15	2 Units, 400 in^3	3.5	7.0*
Fine Sun Sensors	1.5	1 Unit, 6x6x4		
Attitude Control Electronics	12		5	10*
Accelerometer	5	142 in^3	5	
Base Mount, Cover, Wire Harness	27	12x18x32		
G&C Electronics Assembly Support	11			
Coarse Sun Sensor	(2)	8 Units, 3x3x2	None	
Reaction Control Assembly	186.0			
Thrusters & Support	12.6	16 Units, 1.25x1.25x3.0	0	20 (2 roll thrusters)
Nitrogen Tanks	93	4 Units, 15 Dia		
Valves & Regulators	5.2			
Tubing, Brackets & Clamps	13.2			
Nitrogen Gas	62			
Engine Gimbal Actuators				
LMDE	(36)	2 Units	432 (rated)	1152
Limb-Terminator Sensor	(1.4)	2 Units	0.15	

*Both units operating

Table 1.2.2-5: GUIDANCE & CONTROL SUBSYSTEM INTERFACES

ITEM NO.	TYPE OF INTERFACE	INTERFACE DESCRIPTION	INTERFACING SUBSYSTEM	INPUT		BOUNDARY DEFINITION
				TO	FROM	
1	Electrical	Guidance and control subsystem electrical power from buses A, B, and C, 50 volts (peak) 2400 Hz	Power		X	Connectors to Bus A, B, and C
2	Electrical	Guidance and control subsystem electrical power (37-100 Vdc)	Power		X	Cable connectors at G and C
3	Electrical	Input operational command signals, 0 to 6-volt square wave (66 commands)	Computing and Sequencing (DPU)		X	Cable connector at G and C
4	Electrical	Output signals from guidance and control subsystem (12 signals)	Computing and Sequencing	X		Cable connector at G and C
5	Electrical	Input operational command signals, 0 to 6-volt square wave (52 commands)	Computing and Sequencing (CPU)		X	Cable connector at G and C
6	Electrical	Output status signals to telemetry (56 outputs)	Telemetry	X		Cable connector at G and C
7	Electrical	Output to test (27 outputs)	Test Equipment	X		Cable connector at G and C
8	Electrical	Signals to fire squibs in N_2 gas supply (2 supplys)	Pyrotechnic		X	Cable connector at squib valves
9	Mechanical	Provide mechanical support for subsystem components	Structural and Mechanical		X	Mounting surface and fastener pattern holes
10	Thermal	Control subsystem temperatures throughout all ground test and mission phases	Temperature Control		X	Control subsystem temperatures
11	Electrical	Signals actuate solenoid latching valves in N_2 supply for selection, isolation, and cross-tie	Computing and Sequencing		X	Cable connector at G and C
12	Mechanical	Thrust Vector Control Actuator Attach Points (2)	Structure		X	Mounting lug on structure
13	Mechanical	Thrust Vector Control Actuator Attach Points (2)	LMDE	X		Mounting lug on LMDE

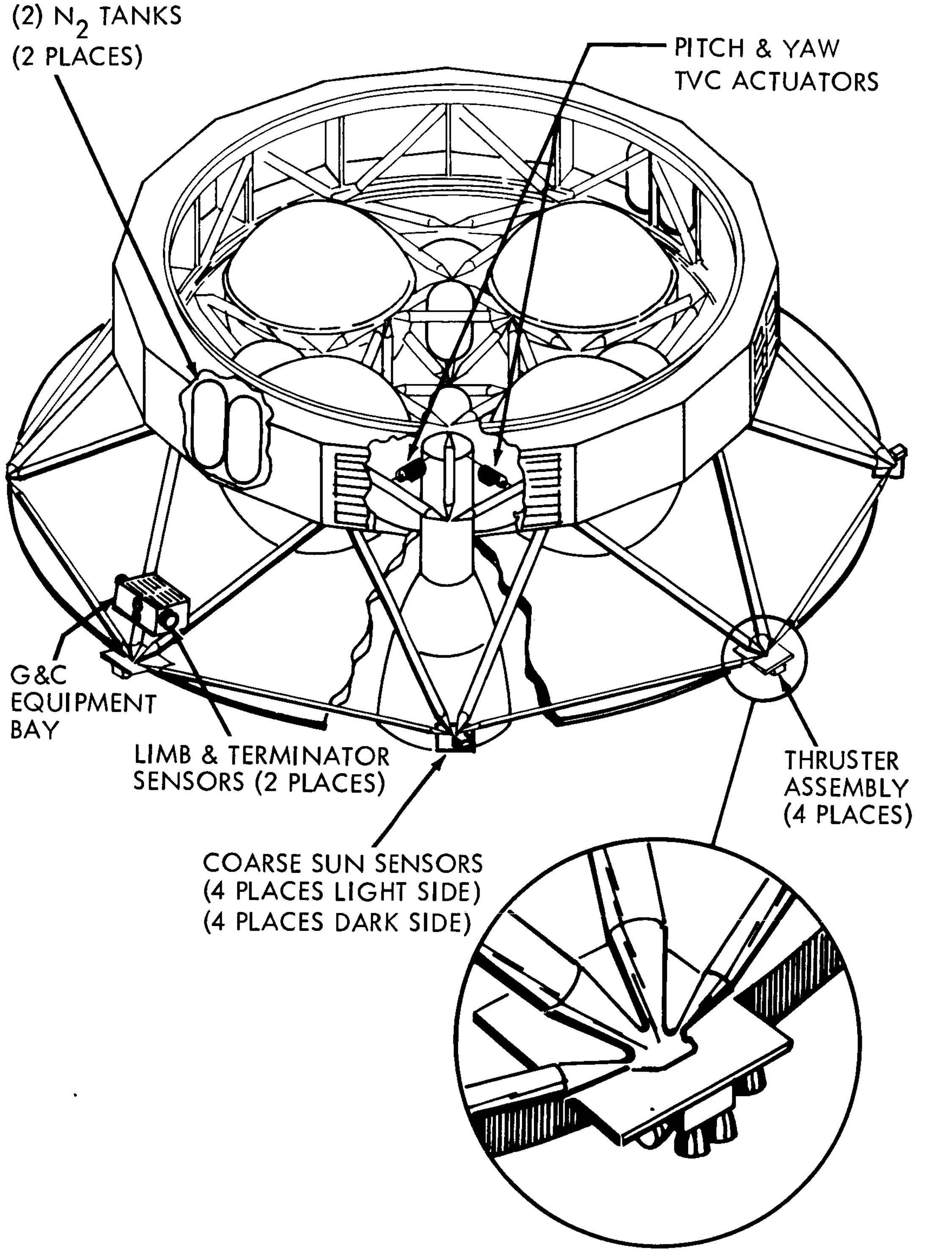


Figure 1.2.2-8: GUIDANCE & CONTROL SUBSYSTEM PHYSICAL INTERFACES

3) Dual cooperative Sun sensors, electronics, and cold-gas systems with dual standby Canopus tracker and gyros.

A single-thread system, as expected, did not meet the reliability requirements.

The other two concepts had comparable reliabilities. The preferred system is the dual standby concept. It was felt that the advantage of no malfunction detection in the cooperative concept was more than offset by the greater weight due to the large propellant redundancy factor required. This change from the Task B preferred concept results from three considerations:

1) The increase in impulse requirements over Task B results in a greater weight penalty for systems having a large redundancy factor.

2) The gas redundancy factor for the cooperative concept was raised from the ideal value of 3 to a more realistic value of 4 as a result of computer studies on the effects of gain mismatch and sensor offset.

3) Greater confidence in the ability to detect failures in the standby concept has been developed as a result of studies conducted since Task B.

In the pitch and yaw axes, the positive and negative error signals are generated and conditioned separately (see Figure 1.2.2-2). This results in a fail-passive system for many possible failures.

Thrust Vector Control Reliability -- Three TVC actuator loop configurations were examined for reliability:

1) Single-thread

2) Standby redundant

3) Parallel cooperative redundant actuator

The gyro contributions were not included in the comparison. Examination of the loop reliabilities revealed that because of the long off-time the dormancy failure rates determined the overall reliabilities. Additionally, the single-thread system showed such high reliability (0.9985) that it was felt that redundancy was neither cost nor weight effective. The preferred design of the TVC loop is therefore single-thread.

1.2.2.6 Trade Study Summary

New trade studies were performed on the guidance and control subsystem and its elements where significant new data have become available or where there have been changes in the subsystem requirements. The preferred design selections were based heavily on considerations for mission success (reliability, performance, proven capability, cost, and weight). Wherever possible, a concerted effort was made to employ a simple, conservative design approach, and to take advantage of design techniques and experience gained in Ranger, Mariner, Lunar Orbiter, and other space programs.

The candidate systems, the selection logic, and the design selections are outlined briefly in Figures 1.2.2-9 through 1.2.2-15.

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE GYRO SELECTION TRADE, MECHANIZATION AND MANEUVER MODE TRADE	SELECTION
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS		MATRIX OF DESIGN APPROACH	

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

REQUIRED GYRO PERFORMANCE

Drift $\leq$ 0.3 deg/hr
Torquing Accuracy - 0.1%
Rate Mode Frequency Response $\geq$ 10 Hz
Noise $\leq$ 20 deg/hr RMS

IRU Requirements

1. Furnish outputs proportional to vehicle pitch, roll, and yaw rates for intial rate damping.
2. Provide means to precisely measure vehicle rates for maneuver control.
3. Measure vehicle pitch, roll, and yaw deviation for inertial hold mode.

MATRIX OF DESIGN APPROACH

1. SINGLE AXIS GYRO

PRO: 1. Well proven, long history of use in space
2. High performance capability.
3. Relatively simple electronics.

CON: 1. Lower reliability - 50,000 hr MTBF.
2. Ball bearing gyros have wearout mechanism.

Redundant single axis gyro IRU mission reliability = 0.987.

2. TWO-AXIS FREE ROTOR GYRO

PRO: 1. Very high demonstrated reliability on Minuteman Program up to 10^6 hrs MTBF
2. Two out of three redundancy
3. No wearout mechanism

CON: 1. Never used in space.
2. Not used to date in strapdown applications
3. Operation in 10 Hz loop not completely verified. Noise may be a problem.
4. Electronics are more complex.

Two-out-of-three redundant free rotor gyro IRU mission reliability = 0.999.

GYRO ELECTRONICS MECHANIZATION

1. DIGITAL

PRO: 1. Single interface with C&S
2. Rate and position available simultaneously.

CON: 1. Complex - D/A conversion required.

Electronics relaibility = 3.2 (10^{-6}) failures/hr

2. ANALOG

PRO: 1. Simple and well proven
2. More reliable than digital mechanization

CON: 1. Only position or rate available unless separate integrator is provided.

Electronics Reliability = 1.7 (10^{-6}) failures/hr

MANEUVER METHODS

1. TORQUE OPEN LOOP GYRO

PRO: 1. Simple, proven.

CON: 1. Derived rate required for convergence.

2. ANALOG INTEGRATION

PRO: 1. Fast maneuver convergence - position and rate available.
2. Gyro temperature control failure is not catastrophic.

CON: 1. Low confidence in stability of operational amplifier integrator.

3. DIGITIZE RATE SIGNAL-INTEGRATE IN C&S

PRO: 1. Simple, proven.
2. Integrated angle available in C&S for direct confirmation by telemetry.

CON: 1. Derived rate required for convergence.

SELECTION

GYRO SELECTION

Reliability 2,1

Confidence 1, 2

Performance 1,2

GYRO ELECTRONICS

Confidence 2, 1

Reliability 2, 1

Performance Comparable

MANEUVER METHODS

Confidence 3, 1, 2

Performance 2, 3, 1

Reliability 1, 3, 2 (all nearly equal)

SELECTED APPRACH

A. Single Axis Gyro

B. Analog Electronics

C. Digitize rate signal

Figure 1.2.2-9: GYRO TRADE STUDY

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE: REACTION CONTROL N_2 SUPPLY ASSEMBLY — MATRIX OF DESIGN APPROACH: SYSTEM WITH SQUIB VALVE.	SYSTEM WITHOUT SQUIB VALVE	SELECTION
FUNCTION AND REQUIRED PERFORMANCE To supply clean nitrogen gas at a regulated pressure to the reaction control thrusters for the purpose of spacecraft attitude control.		1 PRO: 1. Electronic circuitry is less complex, has smaller failure rates and is required to operate during a small portion of the initial phase of the flight. 2. The chance of dirt entering the N_2 supply is less since the tanks are filled only once. 3. The failure rate of parts peculiar to the "with squib" system is 0.289 10^{-6} failures/hour compared to 1.38 x 10^{-6} failures/hour for the "without squib" system. 4. Exposure time of the regulator to high pressure is less. This is especially significant during launch environment. 5. Testing thruster solenoid valves prior to launch will not utilize gas. 6. The design is capable of meeting Voyager sterilization requirements. CON: 1. An auxiliary N_2 supply is required for test and checkout of the thrusters. There is a possibility of jet valve contamination here.	2 PRO: 1. Since the entire system is pressurized from time of assembly, slow leaks can be more easily detected. 2. The total number of parts is less (14 compared to 20 for the system with squibs for a single thread system). 3. The design is capable of meeting Voyager sterilization requirements. CON: 1. A filter must be attached to the system during assembly and checkout to avoid contaminating the N_2 supply. 2. Testing thruster solenoid valves prior to launch will inflict a fuel load penalty. 3. The regulators and thruster valves must be capable of withstanding launch environment while pressurized.	CONFIDENCE 1,2 RELIABILITY 1,2 PERFORMANCE (Equal) SELECTED APPROACH System with squib valve.

Figure 1.2.2-10: NITROGEN SUPPLY

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE NUMBER AND ORIENTATION OF REACTION CONTROL THRUSTERS	SELECTION

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

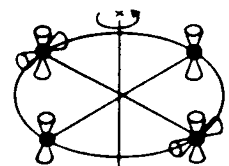
Apply control forces to the spacecraft for attitude control purposes. Arrange control forces so that forces causing perturbations to the trajectory are predictable.

Requirement from JPL Document "Performance and Design Requirements for the 1973 Voyager Mission General Specification For" dated January 1, 1967. Paragraph 3.2.1.1.10, "Planetary Vehicle induced Trajectory Perturbations."

"The unpredictable translational accelerations originating in the Planetary Vehicle shall not exceed a total average value of 0.6×10^{-7} cm/sec^2 (3 σ) time averaged over one hour. The design goal shall be not to exceed 0.2×10^{-7} cm/sec^2 (3σ). This specification is given as the 3σ bounds and is to be interpreted that the actual errors must remain less than the bounds prescribed 99.7% of the time."

MATRIX OF DESIGN APPROACH

1. Use thrusters in couples for all axes.



NUMBER OF THRUSTERS:

12 Thrusters per system
24 required

PERTURBING ACCELERATION TRANS MARS

TRAJECTORY:

$$F = \frac{\text{Impulse Required}}{\text{Time}}$$

$$F = \frac{335 \text{ lb-sec}}{225 \text{ Days}} \times 5\% \text{ Tolerance on Thruster}$$

For limit cycle 87 lb-sec
Disturbance 248 lb-sec

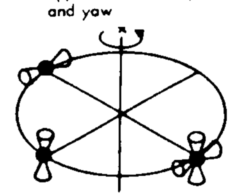
Accel = 0.4×10^{-9} m/sec^2

∴ Thruster tolerances alone are nearly equal to 3σ requirement

PRO: Does not require correction

CON: Degrades reliability, more thrusters, more weight, more power

2. Use couples in roll arrange plus thrusters opposite minus in pitch and yaw



8 Thrusters per system
16 required

Impulse
225 days

248 lb-sec Distrubance

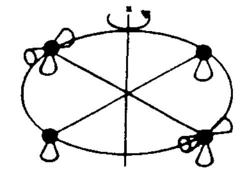
Accel = 0.548×10^{-8} m/sec^2

∴ Correction must account for impulse due to disturbances

PRO: Fewer thrusters, plus and minus impulse balance

CON: Correction required for disturbances

3. Use couples in roll arrange all pitch and yaw thrusters to point toward sun.



8 Thrusters per system
16 required

Total Pitch and Yaw Impulse for Trans Mars
377 lb-sec

Accel = 0.81×10^{-8} m/sec^2

∴ Accuracy of knowledge of impulse must be ≤ 7%

PRO: All pitch and yaw impulse act in known direction.

CON: Accuracy required in propellant budget.

SELECTION

Roll thrusters must be in couples to avoid cross coupling due to relative location in center of gravity.

2 or 3

1 is least but does not meet requirement, therefore select 3 and include capability in software to input impulse.

SELECTED APPROACH

3.

Figure 1.2.2-11: REACTION CONTROL THRUSTERS TRADE STUDY

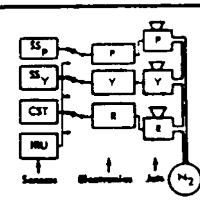
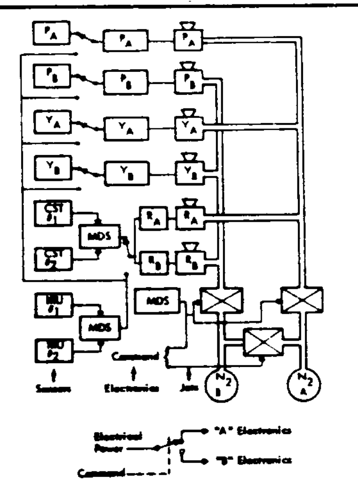
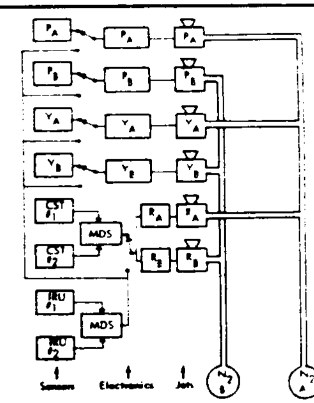
TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE: GUIDANCE AND CONTROL SUBSYSTEM REDUNDANCY CONCEPT			SELECTION
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS		MATRIX OF DESIGN APPROACH: 1 SINGLE THREAD	2 DUAL STANDBY	3 DUAL COOPERATIVE	
Guidance and control subsystem provides spacecraft attitude control throughout the mission. TRADE CONSIDERATIONS 1. Reliability 2. Weight 3. Power 4. Technical Risk		 PRO: 1. Simple 2. Small part count 3. Least weight (propellant quantity factor = 1.5 x basic requirement) CON: 1. Low reliability (0.5952)	 PRO: 1. Good reliability (0.9567) 2. Low weight (propellant quantity factor = 2 x basic requirement) 3. Low power CON: 1. Requires malfunction detection and switching 2. Uncoupled control moment in pitch and yaw.	 PRO: 1. Good reliability (0.9527) 2. No malfunction detection required 3. Control torques are in couples until first failure. CON: 1. Greatest weight (propellant quantity factor = 3 x basic requirement) 2. Greatest power	RELIABILITY 2, 3, 1 SIMPLICITY 1, 3, 2 WEIGHT 1, 2, 3 POWER 1, 2, 3 SELECTED APPROACH Dual Standby (2)

Figure 1.2.2-12: REDUNDANCY CONCEPT TRADE STUDY

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE: REACTION CONTROL PROPELLANT - NITROGEN VS. HYDRAZINE

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

Provide control forces for use in the attitude control of the spacecraft that meet the guidance and control subsystem requirements. Trade items are:

1. Thruster response
2. Minimum impulse bit
3. Reliability
4. Weight
5. Environment of Installation
6. Minimum Power

MATRIX OF DESIGN APPROACH

1

COLD GAS N_2

PRO:

1. Smallest impulse bit: 10 ms and up. "on" delay 3 to 10 ms.
2. Space-proven on:
   Mariner
   Ranger
   Lunar Orbiter
   Surveyor
   Burner II
   etc.
3. Reliability:
   Highest: > 0.93
   Single Thread
4. Thrust Levels:
   As required
5. Thermal Environment
   O.K. From $-65^{\circ}F$ to $+145^{\circ}F$
6. Low power requirements
7. Fixed Thrust Level

CON:

1. Low I_{sp} = 68 sec
   RCS Weight = 185 lb
2. High pressure storage tanks required.

2

HYDRAZINE PLENUM

PRO:

1. Reaction control and plenum now being developed in different systems.
2. Impulse bit:
   20 ms and up, "on" delay
   3 to 10 ms.
3. Medium I_{sp} = 115 sec
   RCS Weight = 73 lb
4. Good prospects for 1977-1979 missions.

CON:

1. No previous application of concept using heat exchanger.
2. Reliability: > 0.92
   Single Thread
3. Thermal Environment:
   Valves and lines limited to $0^{\circ}F$ may need heaters.
   Liquids limit $35^{\circ}F$.
4. High demand rate causes problems with valves, heat sink required.

3

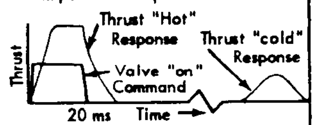
HYDRAZINE ROCKET

PRO:

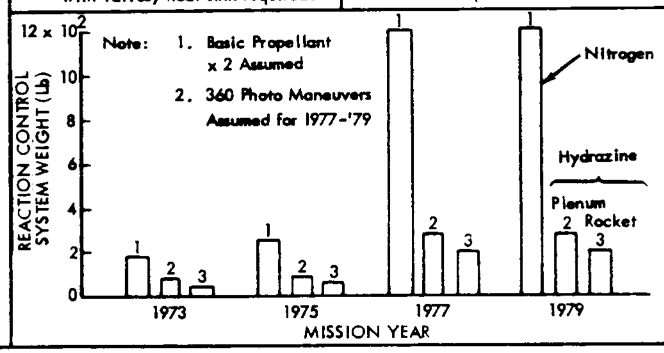
1. Reaction chambers are in development.
2. High I_{sp} = 200 sec
3 RCS Weight = 52 lb
3. Good prospects for 1977-1979 missions.

CONS:

1. Dynamic response not suitable for limit cycle and minimum impulse bit requirements



2. Impulse Bit:
   100 ms
   0.25 lb force & up
3. Reliability:
   Lowest: > 0.90
   Single Thread
4. Heaters required



SELECTION

THRUSTER RESPONSE
1,2,3

IMPULSE BIT
1,2,3

RELIABILITY
1,2,3

WEIGHT
3,2,1

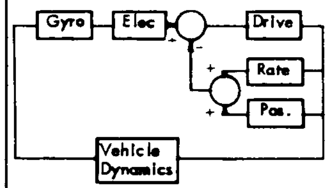
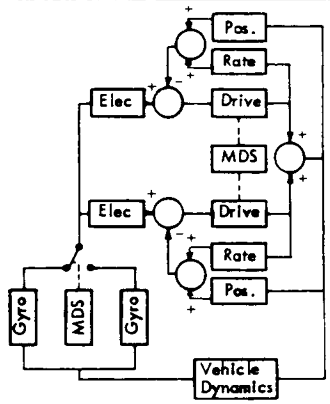
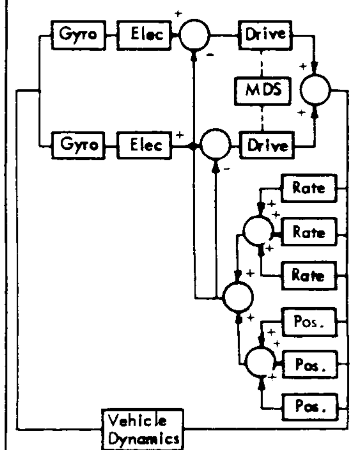
ENVIRONMENT
1,2 and 3

POWER
1,2,3

SELECTED APPROACH

1973 Mission
1.

Figure 1.2.2-13: NITROGEN VS. HYDRAZINE TRADE STUDY

TRADE STUDY SUMMARY SHEET / SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE: THRUST VECTOR CONTROL LOOP REDUNDANCY			SELECTION
	MATRIX OF DESIGN APPROACH			
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	1 SINGLE THREAD	2 STANDBY REDUNDANT	3 PARALLEL COOPERATIVE REDUNDANT	
Thrust vector control loops provide two-axis attitude control during powered flight.				RELIABILITY 3-2, 1 POWER 1-2, 3 WEIGHT 1, 2-3 TECHNICAL RISK 1,2,3
Trade Considerations 1. Reliability 2. Power 3. Weight 4. Technical Risk	PRO: 1. Simple, good reliability (0.9985)* 2. Moderate power requirement (432 watts rated) 3. Least weight 36 lb 4. Least technical risk. Lunar Orbiter technology. CON: 1. Single failure is catastrophic.	PRO: 1. High reliability (0.9990)* Limited by dormancy failures. 2. Similar power to 1. since only single motor runs at one time. (432 watts rated) CON: 1. Higher weight penalty (54 lb) 2. Technical risk in malfunction detection and switching and nonreversing differential drive.	PRO: 1. High Reliability (0.9990)* Limited by dormancy failures. CON: 1. High power consumption (rated 864 watts) 2. Same weight as System 2. 3. High technical risk in obtaining matching performance of components, malfunction detection and switching and nonreversing differential drive.	SELECTED APPROACH 1 Single Thread

*NOTE: All reliability calculations exclude gyro failures since they are included in attitude control.

Figure 1.2.2-14: TVC LOOP REDUNDANCY TRADE STUDY

TRADE STUDY SUMMARY SHEET | **SOURCE OF REQUIREMENT**

TRADE STUDY NUMBER & TITLE: LIMIT-CYCLE DEADBAND SELECTION

MATRIX OF DESIGN APPROACH

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

Without compensation the pitch and yaw limit-cycle deadband amplitude will increase with increasing S/C-Sun distance due to changing Sun sensor scale factor. This effect can be compensated by increasing the position gain as the mission progresses. The alternative is to use a fixed gain set at a value that gives the design deadzone in the vicinity of Mars and a correspondingly smaller deadzone at Earth.

Trade Considerations:

1. Simplicity (Reliability)
2. Flexibility (Versatility)
3. Technical Risk
4. Reaction Control Impulse Requirement

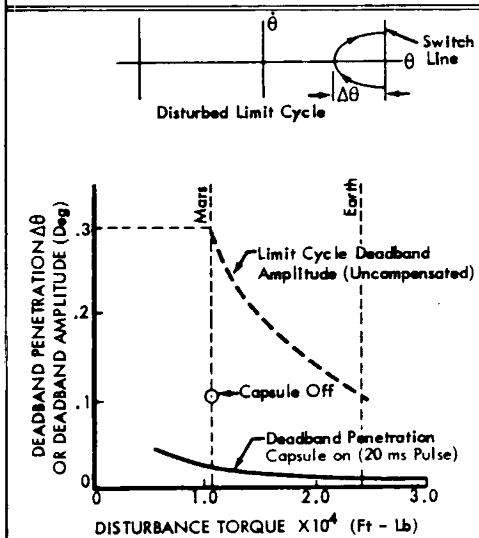
1 VARIABLE GAIN COMPENSATION

PRO:

1. Versatile
   Optional low slew rate.
   Selectable deadzone amplitude.
2. Limit-Cycle deadzone approximates design value throughout mission.
   Low propellant usage for symmetrical (no disturbance) limit cycle.
   Nominal sensor alignment requirements.

CON:

1. Greater complexity in requiring more parts, switch, and possibly command function through C&S.



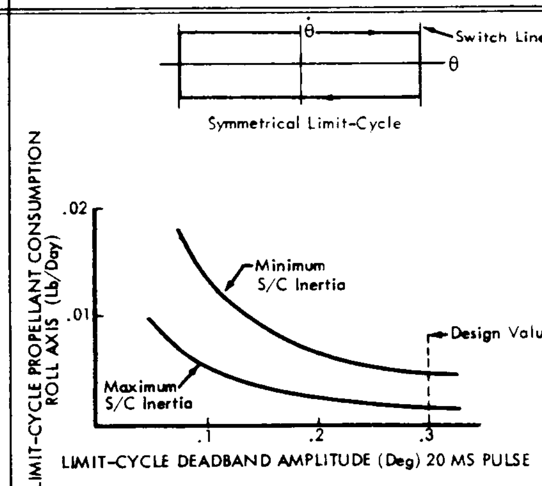
2 FIXED GAIN (NO COMPENSATION)

PRO:

1. Simple (no switching)
2. No propellant penalty if limit-cycle is unsymmetrical due to steady disturbance such as solar pressure on high gain antenna.

CON:

1. Tight deadband at start of mission.
   Sensor alignment and gain are critical over larger region.
   Position gain is higher than needed early in the mission resulting in increased sensitivity to sensor noise.
   Propellant usage will be increased if there is no steady disturbance.



Limit-cycle propellant usage favors largest deadband allowed by antenna pointing and maneuver accuracy requirement which is $\pm$ 0.3 degrees.

Evaluation

1. SIMPLICITY
   2-1
2. FLEXIBILITY
   1-2
3. TECHNICAL RISK
   1-2
4. IMPULSE
   Pitch and Yaw; no penalty for 2
   Roll; not a consideration

SELECTED APPROACH 2

Compensation not required for $\pm$ 0.3 degree deadband at Mars

Figure 1.2.2-15: LIMIT CYCLE DEADBAND TRADE STUDY

1.2.2.7 New Technology and Development Items

Gas Bearing Gyro Tests -- One of the primary advantages of using a gas bearing gyro is that it can be used to back up a malfunctioning optical sensor without fear of gyro wearout. A great deal of confidence has been built up in the use of ball bearing, single-axis gyros in space. Additional testing of gas bearing gyros is recommended to expose any performance and operational anomalies in the gas bearing gyro, and to increase confidence in the gyro reliability.

Following is a list of Boeing experience in single-axis, gas bearing gyro testing:

Gyro	Test Date	Test Type
GG159	June 1965	Low power operation at half speed. Temperature sensitivity, motor vibration capability.
	August 1965	Standard performance tests, noise investigations.
GG334	September 1967	Standard performance tests, temperature sensitivity.

In addition to performance tests, testing is desirable to obtain increased confidence in reliability. The gas bearing gyro has typically a predicted MTBF of 50,000 hours. To prove a 50,000-hour MTBF at an 80% confidence level, at least 150,000 gyro hours must be accumulated. With a test sample of 10 gyros, a 50% confidence level of an MTBF of 50,000 hours would be attained after 1 year of test, which would increase to 85% confidence after 2 years in test, allowing one failure during the test period. The development schedule is shown in Figure 1.2.2-16.

Canopus Sensor Development Studies -- In all previous uses of an image dissector-type Canopus tracker, unforeseen problems have occurred in their operation in space. Notable among these problems have been (1) loss of track due to false targets, (2) stray light problems, and (3) gradual degradation of detector gain with time. Comparative testing of two different Canopus trackers is desirable. The Mariner '69 and Lunar Orbiter Canopus trackers differ in several respects including: (1) S-11 Mariner photocathode versus S-20 Lunar Orbiter, (2) single search-track mode on Mariner versus separate search-track mode on Lunar Orbiter, (3) fiber optics field adapter on Mariner versus lens field adapter on Lunar Orbiter, and (4) electrostatic deflection on Mariner versus electromagnetic deflection on Lunar Orbiter. Different baffle configurations should also be evaluated.

Specific analyses and tests include the following:

1) Design and evaluation of stray light baffles.

2) Evaluation of false target susceptibility. Investigation of methods for minimizing effects of false targets.

3) Evaluation of Canopus discrimination methods.

4) **Investigation and experimental determination of tube gain degradation with time and incident flux density.**

5) Analysis of tracker optical system, including interfaces between aperture, photocathode, field stop, lens system, and baffle system.

A schedule of gyro and Canopus tracker analyses and testing is given in Figure 1.2.2-16.

1.2.2.8 Growth Potential

Growth of the attitude control system can be readily achieved in two major areas:

Malfunction detection and switching

Total impulse capability

The current Task D design calls mainly for ground evaluation of malfunctions and switching between standby redundant units. Development and refinement of a failure modes and effects analysis will lead to more positive definition of the scope of required malfunction detection and switching (MDS). The standby redundancy concept will allow the implementation of on-board MDS operating in parallel with ground-controlled switching.

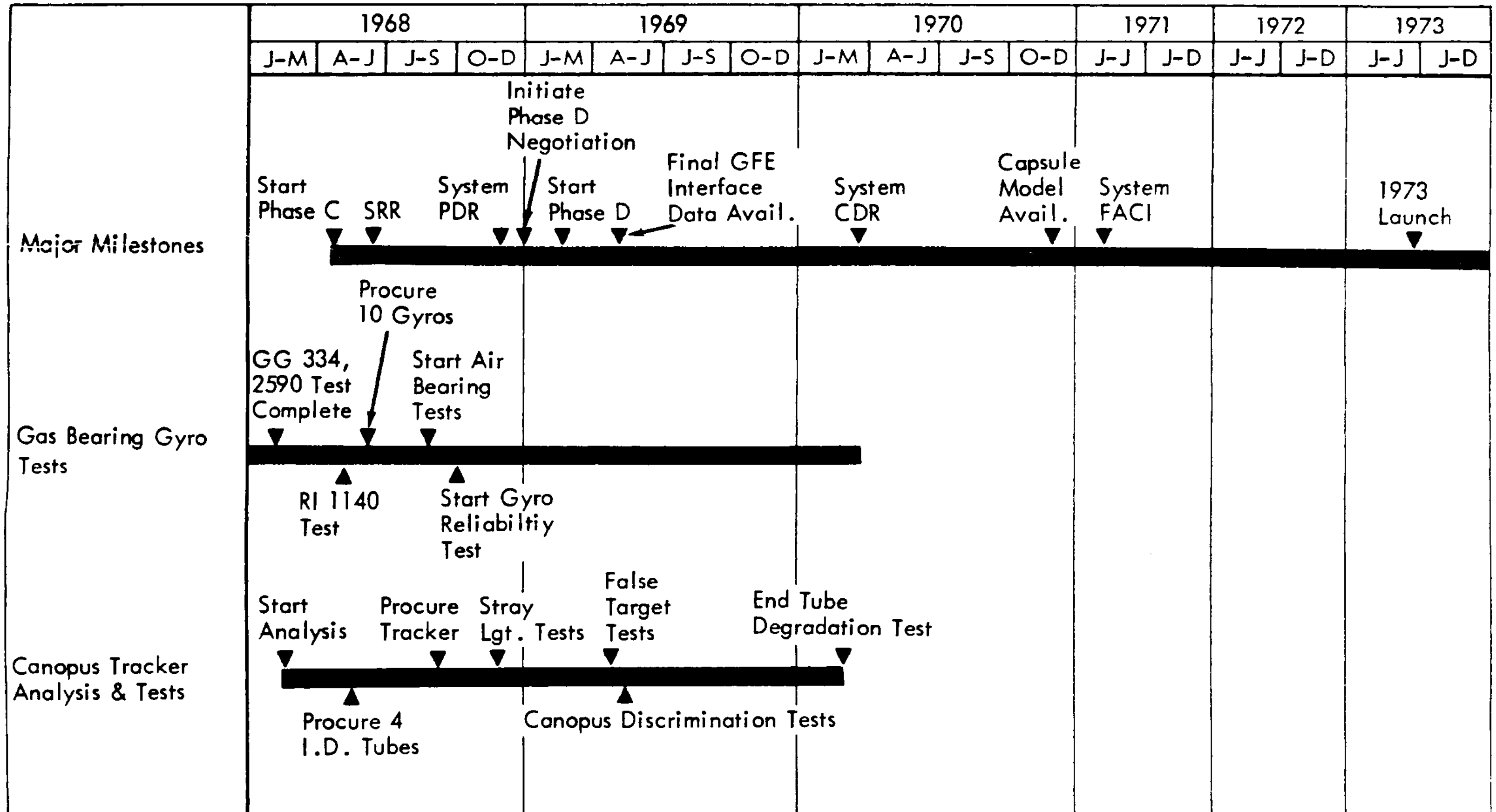
Growth in total impulse requirement can be accommodated by changing the reaction control system propellant from nitrogen to monopropellant hydrazine, or to a hybrid nitrogen and hydrazine system. Operation with a hydrazine plenum concept, operating with gas temperatures less than 250°F, will allow continued use of the cold-gas thrusters. Temperatures generated by gas consumption during maneuvers would be controlled by a heat sink.

Additional total impulse gain can be achieved using a hydrazine primary rocket system. The poor cold-response characteristics of such a system may limit its use to providing maneuvers while limit cycling is provided either by cold nitrogen gas or a hydrazine plenum system. The use of hydrazine introduces a thermal control problem because of the 34°F freezing point of this propellant

Growth to extreme pointing accuracy and stability performance for later mission experiments could be achieved by the introduction of momentum exchange devices probably in the form of control moment gyros. Accuracy and stability would then be limited only by sensor performance.

The thrust vector control system is not expected to require growth for any Mars missions.

Figure 1.2.2-16: GYRO AND CANOPUS TRACKER DEVELOPMENT SCHEDULE



	1968 J-M	1968 A-J	1968 J-S	1968 O-D	1969 J-M	1969 A-J	1969 J-S	1969 O-D	1970 J-M	1970 A-J	1970 J-S	1970 O-D	1971 J-J	1971 J-D	1972 J-J	1972 J-D	1973 J-J	1973 J-D	Total
Labor & Mat'l ($)	68K	300K	300K	200K	66K	30K	25K	25K	25K										1,039K
Capital Facilities ($)	0	0	25K	0	0	0	0	0	0										25K

1.2.3 Telecommunications Link Analysis

1.2.3 Telecommunications Link Analysis

The link performance capability of the command and telemetry modes is established in this section for the Task D design. The relationship of these modes to flight phase as well as to spacecraft and DSIF configuration is presented. The performance of all links is determined through use of design control tables and from supporting analysis for the input parameters. Applicable supporting analysis in the Task A and B reports is referenced; e.g., the performance of the planetary ranging system is essentially the same for the Task D configuration and is only summarized here. This section of the link analysis shows the applicability of the Task A and B analysis, establishes the data mode usage and the spacecraft and DSIF parameters, and gives a performance summary of the telemetry and command modes throughout the mission. The remaining three sections establish the specific performance of the telemetry, command, and ranging modes for each phase of the mission.

Applicability of Task A and B Analysis -- A detailed link analysis and modulation selection was submitted to JPL with the Task A study report and updated for the Task B configuration and data modes. The basic supporting analysis contained in those reports is valid for the Task D design. Included in this category are the selection of carrier and subcarrier modulation techniques, use of a dual-subcarrier baseband, the choice of biorthogonal block (16,5) coding of the upper planetary science channel, and evaluation of receiver tracking performance during the doppler environment of each mission phase.

This present report establishes the data channel power requirements for the new Task D data modes. In addition, the range of performance of each data mode with each equipment configuration is determined.

Flight Sequence and Data Mode Usage -- The proposed flight sequence has been reviewed to establish optimum usage of command, ranging, and telemetry mode configurations provided by the flexibility of the preferred design (Table 1.2.3-1). The more critical links are designated prime links and are subjected to detailed analysis, with the performance of the other links established through appropriate scaling factors.

Spacecraft and DSIF Link Parameters -- Figure 1.2.3-1 shows the pertinent spacecraft RF parameters used in the link analyses. The parameters have been developed in the course of the radio subsystem design presented in Section 1.2.6. The DSIF parameters used in the link analyses are given in Table 1.2.3-2, and are derived primarily from the current issue of EPD-283 (January, 1967).

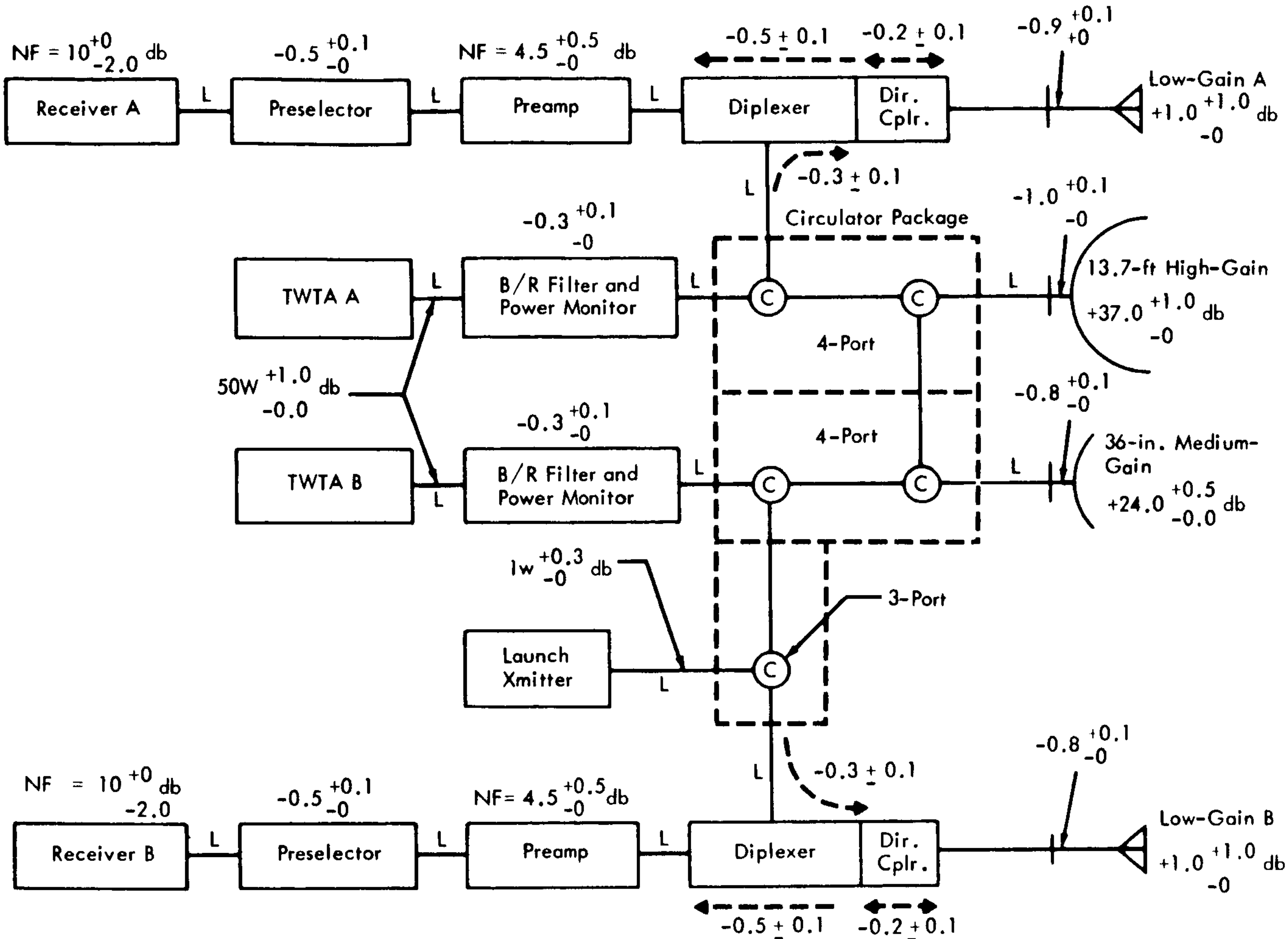
There has been significant change in the DSIF ground antenna gains and system noise temperatures since Task B. For example, the worst-case receive gain of the 210-foot antenna is now given as 60 db, compared to 60.9 db. Likewise, the worst-case system noise temperature using the 210-foot receiving antenna has been redefined to 55°K, compared to the earlier value of 35°K. The degradation of these two DSIF parameters imposes an additional 2.9 db loss factor on all the Task D telemetry links relative to the Task B evaluation.

Performance Summary -- A summary graph (Figure 1.2.3-2) presents launch to end-of-mission command and telemetry performance for the prime links. The capability indicated shows that under worst-case margins the selected links provide adequate performance at specified bit rates and operating periods. The graph is based on the results of the link analyses presented in the following sections. Additional performance capability exists through use of backup modes.

Table 1.2.3-1: TELEMETRY, COMMAND, AND RANGING LINK CONFIGURATIONS BY MISSION PHASE

LINKS ANALYZED			SPACECRAFT			DSN	
Mission Phase	Description	Referenced Link Analysis	Telemetry Mode	Antenna	Power	Antenna	Power
TELEMETRY							
Launch	Hardwired to IU	--	M1	--	1 watt	--	--
Early Cruise	Low Power	--	M1	LGA	1 watt	85 ft	--
*Early Cruise	High Power	Table 1.2.3-4	M1	LGA	50 watts	210 ft	--
Maneuver		--	M1/M6	LGA	50 watts	85/210 ft	--
Late Cruise		--	M1	HGA	50 watts	210 ft	--
*Cruise & Orbit	Emergency	Table 1.2.3-6	M6	LGA	50 watts	210 ft	--
Insertion		--	M2	HGA	50 watts	210 ft	--
*Early Orbit	48,000 bps	Table 1.2.3-5	M2	HGA	50 watts	210 ft	--
Orbit	24,000 bps	--	M3	HGA	50 watts	210 ft	--
Orbit	Backup - MGA	--	M5/M1	MGA	50 watts	210 ft	--
Orbit	Backup-low Power	--	M1	HGA	1 watt	210 ft	--
Late Orbit	12,000 bps	--	M4	HGA	50 watts	210 ft	--
COMMAND							
Parking Orbit	Station 72	--	--	LGA	--	30 ft	10 kw
Postinjection	Acquisition	--	--	LGA	--	2 ft x 2 ft	10 kw
Early Cruise		--	--	LGA	--	85 ft	10 kw
Maneuver	Carrier Only	--	--	LGA	--	85/210 ft	10 kw
*Late Orbit	Prime	Table 1.2.3-8	--	LGA	--	210 ft	10 kw
Late Orbit	Backup	--	--	LGA	--	210 ft	100 kw
RANGING							
Early Cruise	Low Power	--	M1	LGA/LGA	1 watt	85 ft	10 kw
Cruise		--	M1	LGA/LGA	50 watt	85 ft	100 kw
Orbit		--	M1	LGA/HGA	50 watt	210 ft	10 kw

*Links for which design control tables are presented.



Notes:

L = Line Loss = -0.2 ± 0.1 db/Section
All Numbers in db Unless Noted Otherwise

Circulator Package Losses (db)

In \ Out	Low-Gain	Medium Gain	High Gain
TWTA A	$-0.2^{+0}_{-0.2}$	$-0.5^{+0}_{-0.4}$	$-0.3^{+0}_{-0.2}$
TWTA B	$-0.4^{+0}_{-0.4}$	$-0.3^{+0}_{-0.2}$	$-0.5^{+0}_{-0.4}$
L. Xmitter	$-0.2^{+0}_{-0.2}$	$-0.5^{+0}_{-0.4}$	$-0.7^{+0}_{-0.6}$

Total Spacecraft RF Circuit Losses (db)

TWTA A, LGA $-2.5^{+0.7}_{-0.7}$	TWTA A, MGA $-2.2^{+0.5}_{-0.7}$	TWTA A, HGA $-2.2^{+0.5}_{-0.5}$
TWTA B, LGA $-2.6^{+0.7}_{-0.9}$	TWTA B, MGA $-2.0^{+0.5}_{-0.5}$	TWTA B, HGA $-2.4^{+0.5}_{-0.7}$
L.Xmitter, LGA $-1.9^{+0.5}_{-0.6}$	L. Xmitter, MGA $-1.7^{+0.3}_{-0.6}$	L. Xmitter, HGA $-2.1^{+0.3}_{-0.8}$
	LGA to Preamp $-1.7^{+0.4}_{-0.4}$	

Figure 1.2.3-1: SPACECRAFT RF CIRCUIT PARAMETERS

Table 1.2.3-2: DSIF PARAMETERS

PARAMETER	RECEIVE (2295 MHz)	TRANSMIT (2115 MHz)	REFERENCE
Transmitter Power			
85-Foot Sites	--	25/100KW $^{+0.5}_{-0.0}$ db(a)	EPD-283
210-Foot Sites (b) & Stations 72 & 51	--	10KW $^{+0.5}_{-0.0}$ db (a)	EPD-283
Station 71	--	5W $^{+0.5}_{-0.0}$ db	EPD-283
Antenna Gain (c)			
210-Foot	$+61.0 \pm 1.0$ db	$+60.0 \pm 0.8$ db	EPD-283
85-Foot	$+53.0^{+1.0}_{-0.5}$ db	$+51.0^{+1.0}_{-0.5}$ db	EPD-283
30-Foot	$+42.5 \pm 0.5$ db	$+42.0 \pm 0.5$ db	EPD-283
2' x 2' Acquisition	$+22.0 \pm 1$ db	$+19.1 \pm 1.0$ db	EPD-283
4' Station 71	$+25.5^{+1.0}_{-0.5}$ db	$+24.0^{+1.0}_{-1.0}$ db	EPD-283
Antenna Pointing Loss			Computed from angle error of:
210-Foot	-0.1 ± 0.1 db	-0.1 ± 0.1 db	0.02°
85-Foot	$-0.0^{+0.0}_{-0.1}$ db	$-0.0^{+0.0}_{-0.1}$ db	0.02°
30-Foot	-0.5 ± 0.5 db	-0.5 ± 0.5 db	0.3°
4-Foot	$-3.0^{+3.0}_{-1.4}$ db	$-3.0^{+3.0}_{-1.4}$ db	5.0°
2 x 2-Foot	-0.0 ± 0.0 db	-0.0 ± 0.0 db	0.02°
System Noise Temp.			
210- Foot, Maser	45 ± 10°K	--	EPD-283
85- Foot, Maser	55 ± 10°K	--	EPD-283
30- Foot, Paramp	250 ± 50°K	--	EPD-283
4- Foot, Mixer	2642^{+358}_{-609} °K	--	EPD-283
2 x 2 Foot, Paramp	270 ± 50°K	--	EPD-283
Polarization Loss			
S/C HGA—DSIF -1 db Point	$-0.0^{+0.0}_{-0.1}$ db	--	Task B
-3 db Point	$-0.0^{+0.0}_{-0.1}$ db	--	Task B
S/C MGA— DSIF	$-0.2^{+0.0}_{-0.1}$ db	--	Task B
S/C LGA-DSIF	$-0.2^{+0.0}_{-0.1}$ db	$-0.2^{+0.1}_{-0.2}$ db	Task B

(a) Transmitter power is adjustable to 20 db below listed levels.
(b) Planned higher power capability may jeopardize low-noise reception.
(c) Per EPD-283, dated 1 January 1967, antenna gains include line losses.

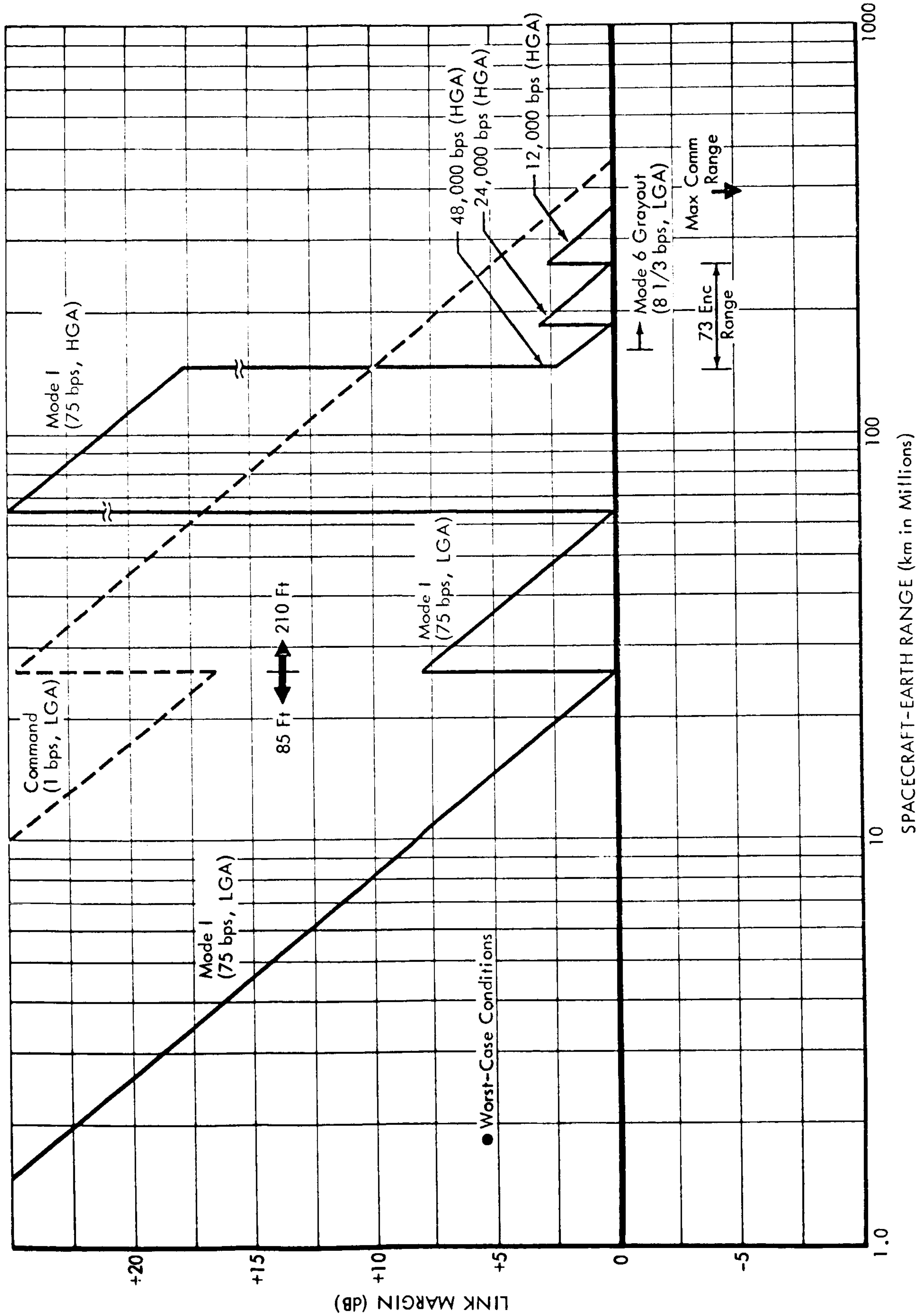


Figure 1.2.3-2: TELEMETRY AND COMMAND LINK PERFORMANCE

1.2.3.1 Spacecraft-to-Earth Telemetry Links

The modulation technique and data modes recommended for the Task D telemetry links are similar to those of the Task B design. The modulation scheme consists of PCM data phase-shift-keyed on sinusoidal subcarriers, which in turn phase-modulate the RF carrier. A single low-data-rate subcarrier is used during launch and cruise (Mode 1), and during emergency backup (Mode 6). During orbital operations, an upper subcarrier using biorthogonal block coding is added to transmit the high-data-rate planetary science and capsule data (Modes 2, 3, 4 and 5). The improvement gained by using the 16,5 biorthogonal block coding was verified in a recent inhouse laboratory research program. The test results of the program, for an information rate of 6000 bps, showed that about a 2-db improvement was provided by the biorthogonal block coding.

The addition of comma freedom to the block coded system for sync purposes is attractive and an experimental program is currently planned to evaluate this feature. An initial analytic evaluation of sequential coding for possible application to Voyager transmission links is also being conducted. It appears that further effort is warranted on both coding alternates.

Table 1.2.3-3 gives the specific link characteristics of each telemetry and command mode in terms of bit rate, subcarrier frequency, tracking bandwidths, threshold signal-to-noise densities, phase deviations, and channel modulation losses. The requirements for bit rate and subcarrier frequencies are contained in the telemetry subsystem description, Section 1.2.5. The receiver tracking bandwidths were determined during Task B. The new required threshold signal-to-noise-density ratios were derived for the higher, Task D, data rates.

The choice of peak phase deviations and the resultant channel modulation losses is based on an optimum power division technique. This technique, described in a recent paper(1), ensures maximum transmission range for each telemetry mode. The phase deviations are chosen such that the individual channels of each data mode threshold simultaneously under worst-case link conditions.

Telemetry Link Performance Analysis -- The specific design control tables developed for the telemetry link modes are discussed in this section. The tables presented were chosen to show the maximum capability of each telemetry data mode identified in Table 1.2.3-1, operating with the various anticipated spacecraft-DSIF component combinations. Telemetry link performance for all links has been determined, although only prime-link design control tables are presented here. For links where design control tables are not presented, performance capability is summarized.

For each design control table, a performance graph shows the limiting channel and indicates relative margins of the other channels. A limiting channel is defined as the channel which first enters "grayout", where the grayout period starts at the point where nominal performance margins become equal to worst-case tolerances.

Performance and Design Control Tables for Primary Links -- The flexibility of the Task D preferred design allows many combinations of spacecraft components and telemetry data modes. In addition, as the flight progresses from launch through acquisition, cruise, encounter, and orbit phases, several different DSIF capabilities are available.

(1) T. K. Foley, B. J. Gaumond, and J. T. Witherspoon, "Optimum Power Division for Phase-Modulated Deep Space Communications Links", IEEE Transactions on Aerospace and Electronic Systems, Vol. AES-3, No. 3, May 1967

MODE	CHANNEL	SUBCARRIER MODULATION	SUBCARRIER FREQUENCY, Hz	BIT RATE		BIT ERROR RATE, P_e^b	REQ'D ST/N/B, db	LOOP $2B_{LO}$		REQ'D $(S/N)_{2B_{LO}}$, db	WORST-CASE CHANNEL THRESHOLD SIGNAL-TO NOISE DENSITY, db	PEAK PHASE DEVIATION, RADS.			CARRIER MOD. LOSS, db	SYNC OR LOWER CHANNEL MOD. LOSS, db	UPPER CHANNEL MOD. LOSS, db
				bps	db			Hz	db			MIN (-10%)	NOM	MAX (+10%)			
1	Carrier	--	--	--	--	--	--	48 12 5	+16.8 +10.8 +7.0	+15.0	+31.8 +25.8 +22.0	--	--	--	$-5.4^{+1.2}_{-1.4}$		
	Data	Coherent PSK	1200	75	+18.8	5×10^{-3}	$+7.9\pm0.3$	--	--	--	+27.0	1.305	1.45	1.595		$-2.2^{+0.3}_{-0.4}$	--
2	Carrier	--	--	--	--	--	--	12	+10.8	+20.0	+30.8	--	--	--	$-5.3^{+1.2}_{-1.3}$		
	Lower	Coherent PSK	4800	300	+24.8	5×10^{-3}	$+7.2\pm0.3$	--	--	--	+32.3	0.297	0.33	0.363		$-17.8^{+1.9}_{-2.2}$	
	Upper	Biorth. (16,5) Coded Coherent PSK	614,400	48,000	+46.8	5×10^{-3}	$+5.2\pm0.3$	--	--	--	+52.3	1.296	1.41	1.551			$-2.5^{+0.3}_{-0.5}$
3	Carrier	Same as Mode 2	--	--	--	--	--	12	+10.8	+20.0	+30.8	--	--	--	$-5.3^{+1.2}_{-1.3}$		
	Lower		2400	300	+24.8	5×10^{-3}	$+7.2\pm0.3$	--	--	--	+32.3	0.297	0.33	0.363		$-17.8^{+1.9}_{-2.2}$	
	Upper		614,400	24,000	+43.8	5×10^{-3}	$+5.2\pm0.3$	--	--	--	+49.3	1.296	1.41	1.551			$-2.5^{+0.3}_{-0.5}$
4	Carrier		--	--	--	--	--	12	+10.8	+20.0	+30.8	--	--	--	$-5.3^{+1.1}_{-1.3}$		
	Lower	Same as Mode 2	1200	300	+24.8	5×10^{-3}	$+7.2\pm0.3$	--	--	--	+32.3	0.396	0.44	0.484		$-15.2^{+1.8}_{-2.2}$	
	Upper		614,400	12,000	+40.8	5×10^{-3}	$+5.2\pm0.3$	--	--	--	+46.3	1.251	1.39	1.526			$-2.8^{+0.4}_{-0.6}$
5	Carrier		--	--	--	--	--	12	+10.8	+20.0	+30.8	--	--	--	$-5.1^{+1.0}_{-1.2}$		
	Lower	Same as Mode 2	1200	120	+20.8	5×10^{-3}	$+7.2\pm0.3$	--	--	--	+28.3	0.621	0.69	0.759		$-10.8^{+1.6}_{-1.8}$	
	Upper		61,440	1200	+30.8	5×10^{-3}	$+5.2\pm0.3$	--	--	--	+36.3	1.152	1.28	1.408			$-3.8^{+0.6}_{-0.8}$
6	Carrier	--						5	+7.0	+9.0	+16.0	--	--	--	$-4.6^{+0.9}_{-1.1}$		
	Sync	2-Channel Coherent PSK	75	--	--	--	--	$0.5^{+0.05}_{-0.04}$	-3.0 ± 0.4	$+12.5\pm0.3$	+10.2	0.486	0.54	0.594		$-10.0^{+1.4}_{-1.5}$	
	Data		150	8 1/3	+9.2	5×10^{-3}	$+6.8\pm0.3$	--	--	--	+16.3	1.052	1.17	1.287			-4.6 ± 0.8
Command	Carrier	--	--	--	--	--	--	20 ± 2	$+13.0^{+0.4}_{-0.5}$	+9.0	+22.4	--	--	--	-1.4 ± 0.3		
	Sync		--	--	--	--	--	$0.4^{+0}_{-0.08}$	$-4.0^{+0}_{-1.0}$	+18.7	+14.7	0.414	0.46	0.506		$-8.4^{+0.8}_{-1.0}$	
	Data		--	1	0.0	1×10^{-5}	$+11.7^{+0}_{-1.0}$	--	--	--	+11.7	0.396	0.44	0.484			$-11.3^{+1.0}_{-1.1}$

Table 1.2.3-3: TELEMETRY AND COMMAND MODULATION PARAMETERS

1) Telemetry Mode 1 -- Mode 1 is the primary data mode used during all flight phases before Mars encounter. This mode transmits cruise science data, capsule data, tape recorder playback data, and spacecraft engineering data at the rate of 75 bits per second. Table 1.2.3-1 gives the various spacecraft and ground equipment configurations used with Mode 1 at different flight phases.

In the Task B design, Mode 1 telemetry at 80 bps was transmitted during launch through an RF window in the shroud with a backup through the Saturn V instrument unit (IU). For Task D, launch phase telemetry is by hardline from the planetary vehicles to the Saturn V IU for subsequent transmission. During this time period, the spacecraft S-band carrier signal is coupled into the IU S-band transponder antenna for transmission.

Table 1.2.3-4 is the detailed design control table used to establish Mode 1 performance during the mid-cruise period. The spacecraft uses the 50-watt transmitter and low gain antenna to transfer 75 bps to the 210-foot DSIF sites. Figure 1.2.3-3 is the resultant performance graph for Mode 1. The graph also includes the results of analyzing Mode 1 performance for early cruise (low gain antenna/85 ft) and late cruise (high gain antenna/210 ft). The performance evaluation shows that, under worst case, Mode 1 via the low-gain antenna operates to a range of 26×10^6 km with the 85-foot DSIF network. This point corresponds to 48 days from launch for a cruise trajectory having a late, April 4, 1974, encounter. Switchover to the 210-foot DSIF sites permits Mode 1 operation to 64×10^6 km, worst case, before requiring use of the spacecraft high-gain antenna. This switchover is 91 days from launch for an April 4th encounter. Substantial margins are provided using the high-gain antenna, and a 20-db margin worst case is available at the maximum communication range (393×10^6 km). With nominal performance margin, the above switchovers would occur at 39×10^6 km (66 days) and 95×10^6 km (115 days), respectively.

2) Telemetry Mode 2 -- Mode 2 is the primary early-orbit-phase data mode, transmitting 48,000 bps of planetary science data on an upper subcarrier, and 300 bps of combined engineering and cruise science data on a lower subcarrier. Design control Table 1.2.3-5 presents the link parameters and calculations used for evaluation. The performance graph (Figure 1.2.3-4) shows that Mode 2 operates to a range of 185×10^6 km before entering grayout (worst-case margins), and to a range of 300×10^6 km under nominal link performance, before entering blackout. The total time of operation in Mode 2 (48,000 bps) depends on the encounter date. For the case of early encounter (143×10^6 km), the mode operates for about 3 weeks before grayout and an additional 3 months in the grayout region. In contrast, for a late encounter (260×10^6 km), the mode operates, entirely in the greyout region, for only 1 month under nominal margins before blackout.

3) Telemetry Modes 3 and 4 -- Modes 3 and 4 are the mid- and late-orbit data modes. Mode 3 transmits 24,000 bps and Mode 4 transmits 12,000 bps of planetary science data on the upper subcarrier; each mode also transmits 300 bps of data on the lower channel as in Mode 2. The performance of each mode was evaluated using the modulation parameters of Table 1.2.3-1 in design control tables similar to that of Mode 2 (Table 1.2.3-5). The resultant performance capability and mode switchover points are shown in Figure 1.2.3-5 for Modes 2, 3, and 4, the prime orbital modes. The performance graph shows

Table 1.2.3-4: DESIGN CONTROL TABLE - MODE 1 (CRUISE)

PROJECT: VOYAGER 1973, TASK D DATE:

CHANNEL: SPACECRAFT-EARTH TELEMETRY LINK, MODE 1 (75 bps)

MODE: SPACECRAFT - 50W, LGA; DSIF -210 ft., MASER

NO.	PARAMETER	VALUE	TOLERANCE	SOURCE
1	Total Transmitter Power, 50W	+ 47.0 dbm	+1.0, -0.0 db	Fig. 1.2.3-1
2	Transmitting Circuit Loss	- 2.5 db	+0.7, -0.7 db	Fig. 1.2.3-1
3	Transmitting Antenna Gain, LGA	+ 1.0 db	+1.0, -0.0 db	Fig. 1.2.3-1
4	Transmitting Antenna Pointing Loss	Included in (3)	---	
5	Space Loss (2295 mc, 50×10^6 km)	-253.7 db	---	
6	Polarization Loss	- 0.2 db	+0.0, -0.1 db	Table 1.2.3-2
7	Receiving Antenna Gain, 210 ft.	+ 61.0 db	+1.0, -1.0 db	Table 1.2.3-2
8	Receiving Antenna Pointing Loss	- 0.1 db	+0.1, -0.1 db	Table 1.2.3-2
9	Receiving Circuit Loss	Included in (7)	---	
10	Net Circuit Loss	-194.5 db	+2.8, -1.9 db	
11	Total Received Power	-147.5 dbm	+3.8, -1.9 db	
12	Receiver Noise Spectral Density (N/B) (T System = 45, ±10 °K)	-182.1 dbm	+0.9, -1.1 db	Table 1.2.3-2
13	Carrier Modulation Loss	- 5.4 db	+1.2, -1.4 db	Table 1.2.3-3
14	Received Carrier Power	-152.9 dbm	+5.0, -3.3 db	
15	Carrier APC Noise Bw ($2B_{LO}$ = 5 Hz)	+ 7.0 db	+0.0, -0.5 db	Table 1.2.3-3
	CARRIER PERFORMANCE — TRACKING (ONE WAY)			
16	Threshold SNR in $2B_{LO}$	+ 9.0 db	---	Table 1.2.3-3
17	Threshold Carrier Power	-166.1 dbm	+0.9, -1.6 db	
18	Performance Margin	+ 13.2 db	+6.6, -4.2 db	
	CARRIER PERFORMANCE — TRACKING (TWO WAY)			
19	Theshold SNR in $2B_{LO}$	+ 9.0 db	---	Table 1.2.3-3
20	Threshold Carrier Power	-166.1 dbm	+0.9, -1.6 db	
21	Performance Margin	+ 13.2 db	+6.6, -4.2 db	
	CARRIER PERFORMANCE — DATA DEMODULATION			
22	Threshold SNR in $2B_{LO}$	+ 15.0 db	---	Table 1.2.3-3
23	Threshold Carrier Power	-160.1 dbm	+0.9, -1.6 db	
24	Performance Margin	+ 7.2 db	+6.6, -4.2 db	
	DATA CHANNEL			
25	Modulation Loss	- 2.2 db	+0.3, -0.4 db	Table 1.2.3-3
26	Received Data Subcarrier Power	-149.7 dbm	+4.1, -2.3 db	
27	Bit Rate (1/T), 75 bps	+ 18.8 db	---	Table 1.2.3-3
28	Required ST/N/B	+ 7.9 db	+0.3, -0.3 db	Table 1.2.3-3
29	Threshold Subcarrier Power	-155.4 dbm	+1.2, -1.4 db	
30	Performance Margin	+ 5.7 db	+5.5, -3.5 db	
	DATA CHANNEL			
31	Modulation Loss			
32	Received Data Subcarrier Power			
33	Bit Rate (1/T)			
34	Required ST/N/B			
35	Threshold Subcarrier Power			
36	Performance Margin			

COMMENTS:

Link will operate above 0-db margin to range of 64.5×10^6 km worst case, 96.5×10^6 km nominal.

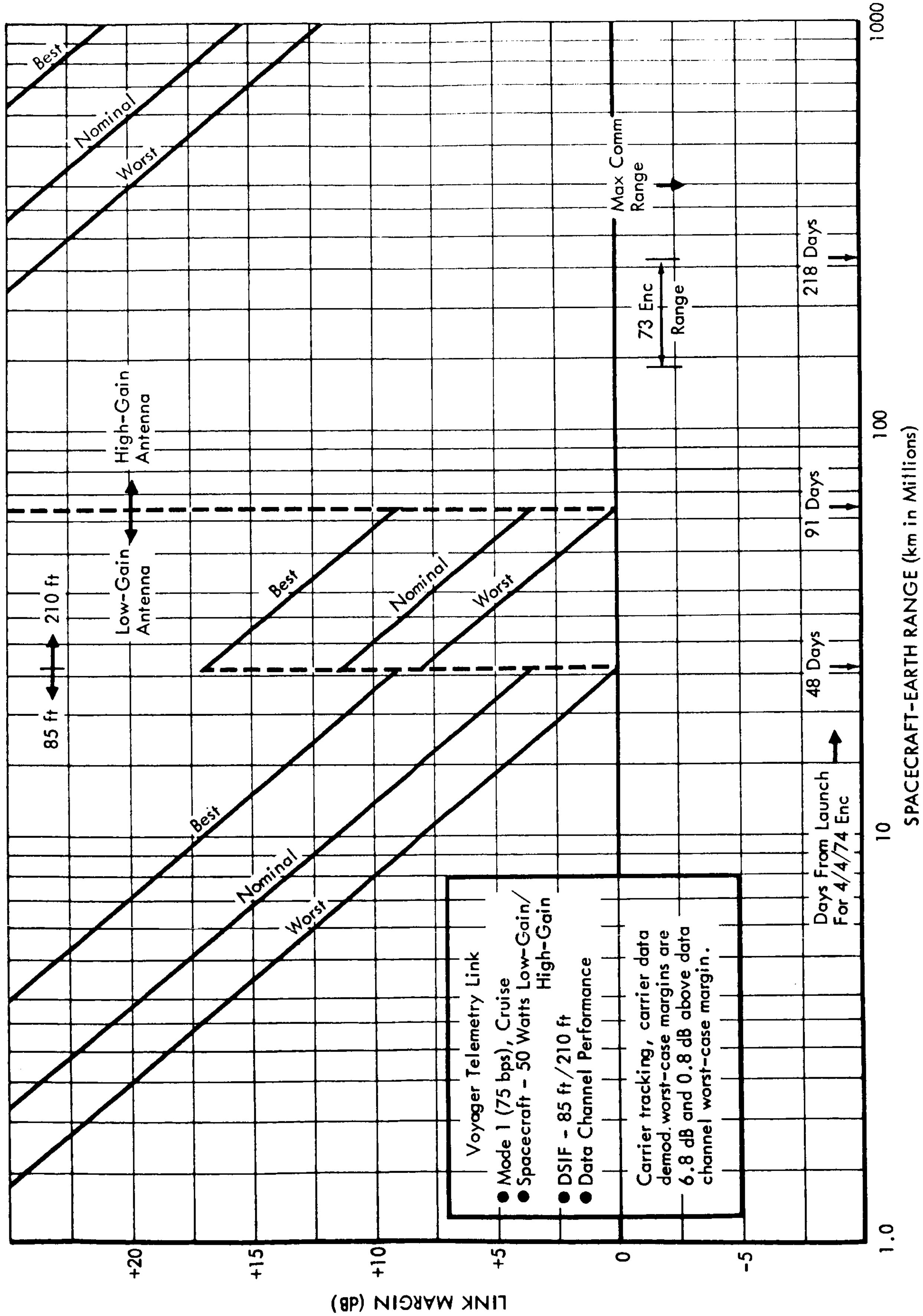


Figure 1.2.3-3: MODE 1 PERFORMANCE

Table 1.2.3-5: DESIGN CONTROL TABLE - MODE 2

PROJECT: VOYAGER 1973, TASK D DATE: 9/8/67

CHANNEL: SPACECRAFT EARTH TELEMETRY LINK, MODE 2 (48,000/300 bps)

MODE: SPACECRAFT - 50W, HGA; DSIF - 210 FT, MASER

NO.	PARAMETER	VALUE	TOLERANCE	SOURCE
1	Total Transmitter Power, 50W	+ 47.0 dbm	+1.0, -0.0 db	Fig. 1.2.3-1
2	Transmitting Circuit Loss	- 2.2 db	+0.5, -0.5 db	Fig. 1.2.3-1
3	Transmitting Antenna Gain, HGA	+ 37.0 db	+1.0, -0.0 db	Fig. 1.2.3-1
4	Transmitting Antenna Pointing Loss	- 1.0 db	+1.0, -0.9 db	Sec. 1.2.7.2.2)
5	Space Loss (2295 mc, 200 x 10^6 km)	-265.7 db	---	
6	Polarization Loss	0.0 db	+0.0, -0.1 db	Table 1.2.3-2
7	Receiving Antenna Gain, 210 Ft.	+ 61.0 db	+1.0, -1.0 db	Table 1.2.3-2
8	Receiving Antenna Pointing Loss	- 0.1 db	+0.1, -0.1 db	Table 1.2.3-2
9	Receiving Circuit Loss	Included in (7)	---	
10	Net Circuit Loss	-171.0 db	+3.6, -2.6 db	
11	Total Received Power	-124.0 dbm	+4.6, -2.6 db	
12	Receiver Noise Spectral Density (N/B) (T System = 45, ±10 °K)	-182.1 dbm	+0.9, -1.1 db	Table 1.2.3-2
13	Carrier Modulation Loss	- 5.3 db	+1.2, -1.3 db	Table 1.2.3-3
14	Received Carrier Power	-129.3 dbm	+5.8, -3.9 db	
15	Carrier APC Noise Bw ($2B_{LO}$ = 12 Hz)	+ 10.8 db	+0.0, -0.5 db	Table 1.2.3-3

CARRIER PERFORMANCE — TRACKING (ONE WAY)

16	Threshold SNR in $2B_{LO}$	+ 9.0 db	---	Table 1.2.3-3
17	Threshold Carrier Power	-162.3 dbm	+0.9, -1.6 db	
18	Performance Margin	+ 33.0 db	+7.4, -4.8 db	

CARRIER PERFORMANCE — TRACKING (TWO WAY)

19	Theshold SNR in $2B_{LO}$	+ 9.0 db	---	Table 1.2.3-3
20	Threshold Carrier Power	-162.3 dbm	+0.9, -1.6 db	
21	Performance Margin	+ 33.0 db	+7.4, -4.8 db	

CARRIER PERFORMANCE — DATA DEMODULATION

22	Threshold SNR in $2B_{LO}$	+ 20.0 db	---	Table 1.2.3-3
23	Threshold Carrier Power	-151.3 dbm	+0.9, -1.6 db	
24	Performance Margin	+ 22.0 db	+7.4, -4.8 db	

DATA CHANNEL - LOWER

25	Modulation Loss	- 17.8 db	+1.9, -2.2 db	Table 1.2.3-3
26	Received Data Subcarrier Power	-141.8 dbm	+6.5, -4.8 db	
27	Bit Rate (1/T), 300 bps	+ 24.8 db	---	Table 1.2.3-3
28	Required ST/N/B	+ 7.2 db	+0.3, -0.3 db	Table 1.2.3-3
29	Threshold Subcarrier Power	-150.1 dbm	+1.2, -1.4 db	
30	Performance Margin	+ 8.3 db	+7.9, -6.0 db	

DATA CHANNEL - UPPER

31	Modulation Loss	- 2.5 db	+0.3, -0.5 db	Table 1.2.3-3
32	Received Data Subcarrier Power	-126.5 dbm	+4.9, -3.1 db	
33	Bit Rate (1/T), 48,000 bps	+ 46.8 db	---	Table 1.2.3-3
34	Required ST/N/B	+ 5.2 db	+0.3, -0.3 db	Table 1.2.3-3
35	Threshold Subcarrier Power	-130.1 dbm	+1.2, -1.4 db	
36	Performance Margin	+ 3.6 db	+6.3, -4.3 db	

COMMENTS:

Link will operate above 0-db margin to range of 185 x 10^6 km worst case, 303 x 10^6 km nominal.

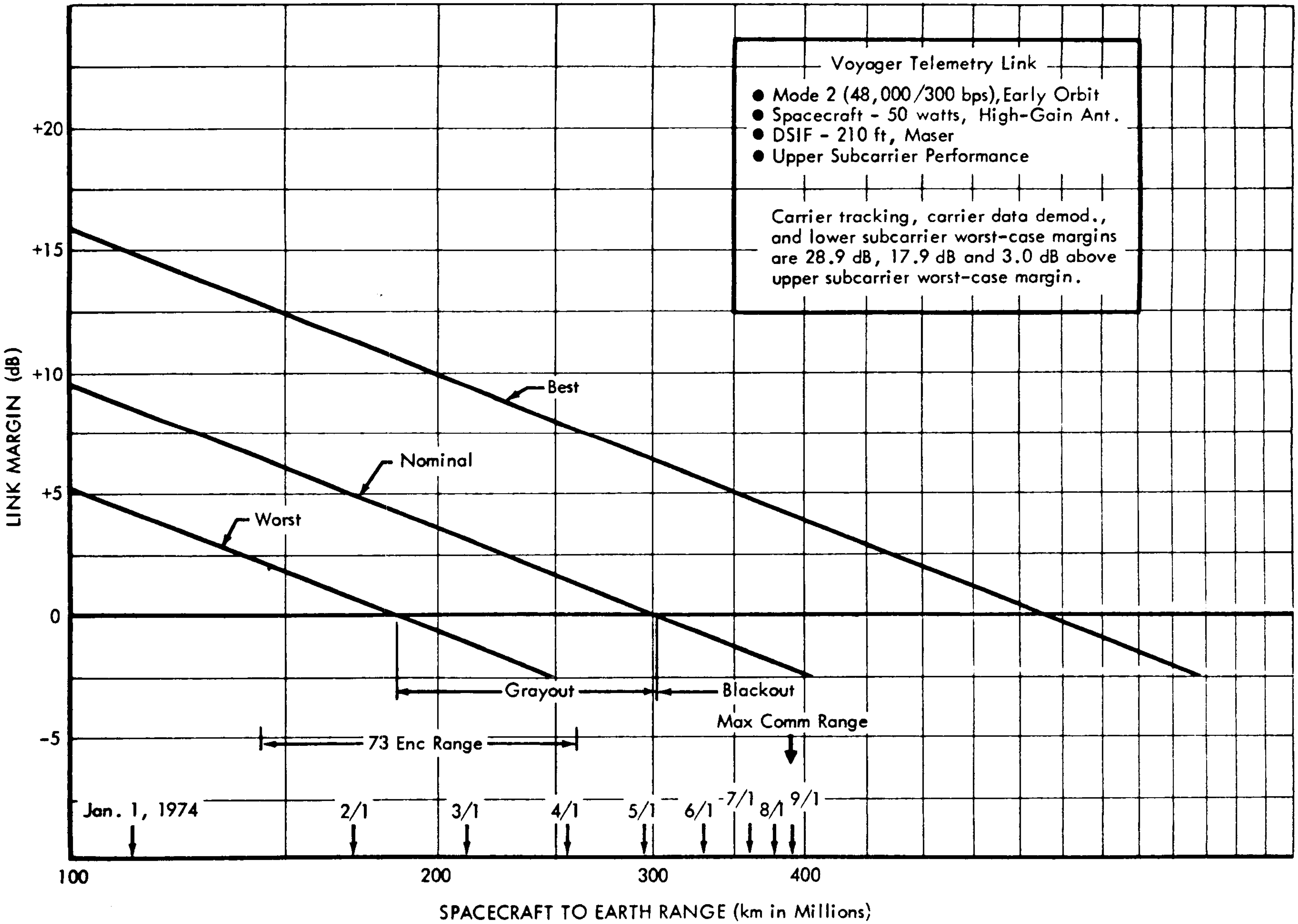


Figure 1.2.3-4: MODE 2 PERFORMANCE

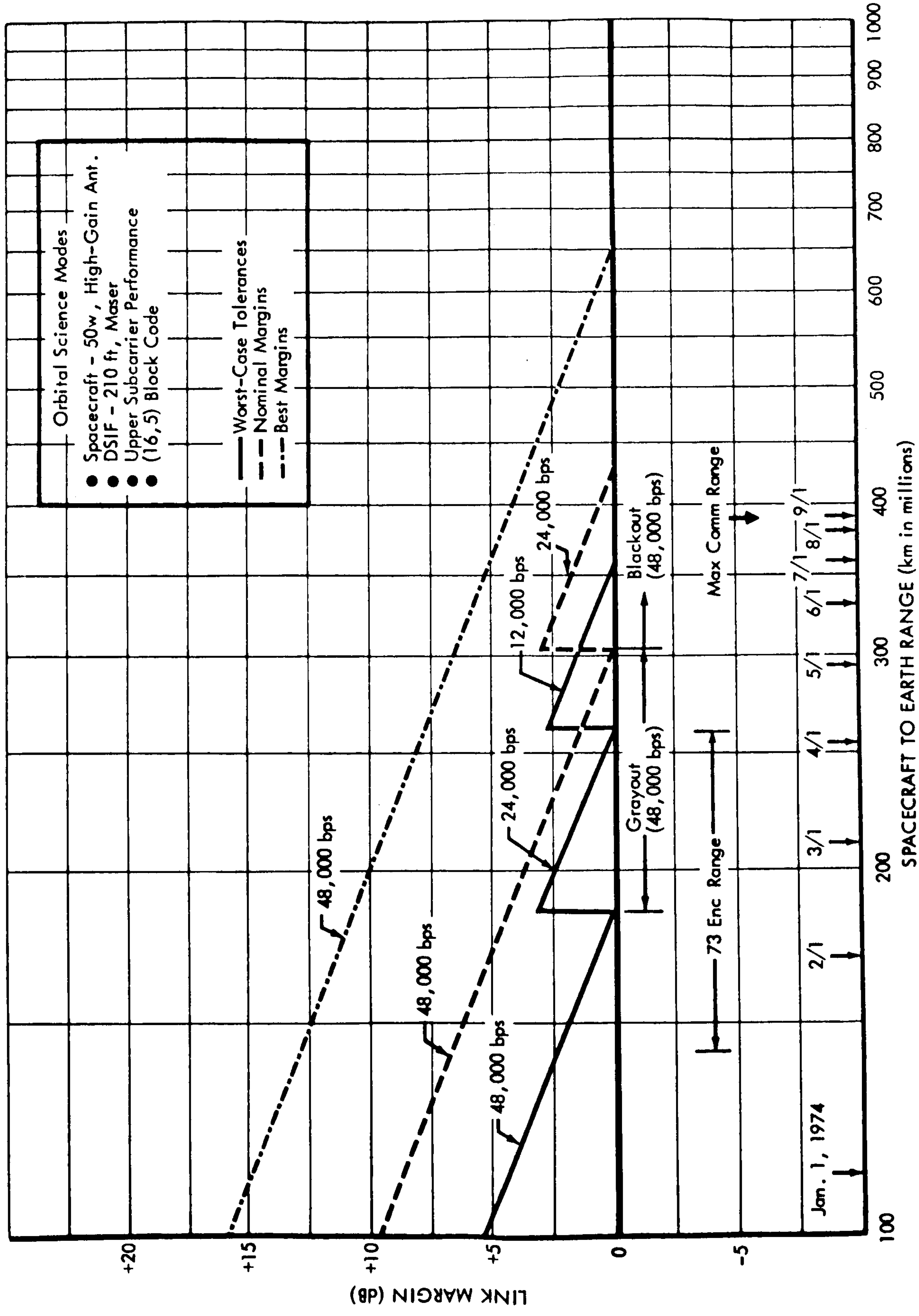


Figure 1.2.3-5: ORBITAL LINK PERFORMANCE

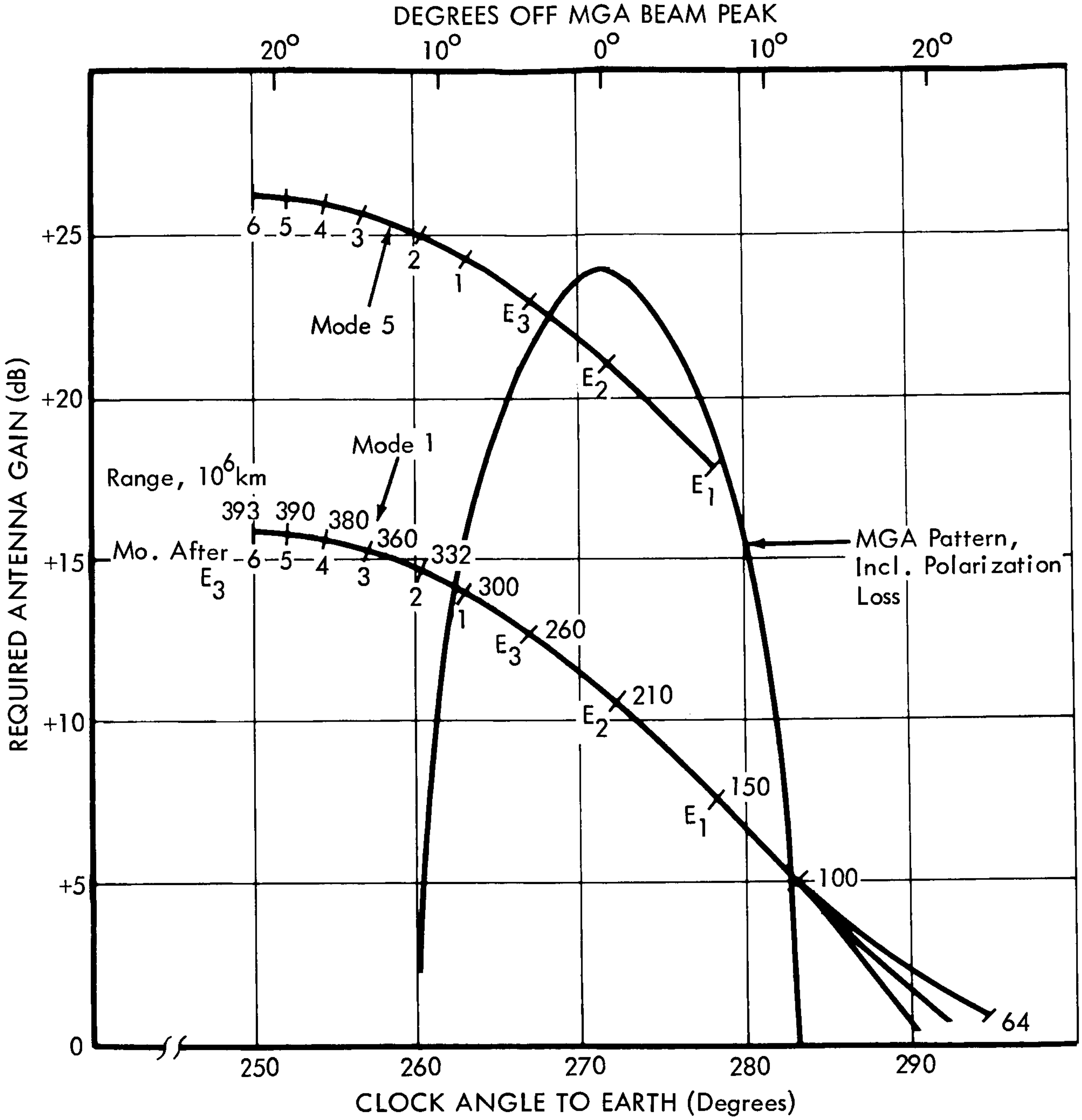
that, under worst-case tolerances, Mode 2 thresholds at 185 x 10^6km. Mode 3 is then activated and operates to a range of 261x10^6km, at which point Mode 4 is initiated and operates to 352x10^6km. Under nominal margin conditions, Mode 2 (48,000 bps) operates to 300x10^6km; Mode 3 is then activated and operates above threshold beyond the maximum communication range of 393x10^6km.

The monthly time scale on the graph permits evaluation of the operation time in each mode for different encounter dates and link threshold criteria. For example, an early encounter (143x10^6km) and worst-case link margins provide communication for 1 month at 48,000 bps, plus 1.7 months at 24,000 bps, and 2.7 months at 12,000 bps, for a total planetary science data return of 3.14x10^{11} bits. The link operating under nominal margins for a 6-month mission and the same early encounter provides 48,000 bps for 3.8 months and 24,000 bps for the final 2.2 months. These rates provide a total planetary science data return of 5.97x10^{11} bits. Similarly, if an extremely late encounter (260x10^6km) is examined under worst-case and nominal conditions, the following results are obtained: (a) under worst-case margins, the link operates in Mode 4 (12,000 bps) for about 2.7 months, returning 0.84x10^{11} bits of planetary science data; (b) under nominal margins, Mode 2 (48,000 bps) is used for 1.1 month and Mode 3 (24,000 bps) for the following 4.9 months to return 4.42x10^{11} total bits of planetary science data for a 6-month mission. If the spacecraft operates beyond 6 months in orbit, the communication range decreases from the maximum, and higher total data return can be expected.

4) Mode 5 Performance -- A 36-inch medium gain antenna, described in Section 1.2.7.2, provides backup capability to the high gain antenna during the early orbit mission phase. Mode 5 transmission permits 1200/120 bps of data over the medium gain antenna, thus providing significant science data return even in the event of high gain antenna failure. The medium gain antenna has single-axis drive, and is positioned in its fixed axis prior to launch to provide maximum length of coverage beyond the selected encounter date.

The ecliptic plane is chosen for the scanning plane of the medium gain antenna, since this choice both maximizes Mode 5 communication time after encounter and permits an additional Mode 1 backup capability before encounter. The positioning of the fixed axis of the medium gain antenna, out of the ecliptic plane, is a function of the encounter date. Figure 1.2.3-6 illustrates the fixed axis position selection. Required medium gain antenna gain, a function of range, is plotted for both Modes 1 and 5 versus the clock angle to Earth, which is a function of the spacecraft Earth cruise and orbit geometry. The medium gain antenna pattern, having a peak gain of 24.0 db, is then slid along the clock angle axis of the graph to determine the points of best coverage for a specified encounter range. The link is realizable whenever the medium gain antenna pattern gain is higher than the required antenna gain for the particular mode under consideration. Figure 1.2.3-6 shows the optimum medium gain antenna positioning for an early encounter, at a range of 150x10^6km. The Mode 5 performance for early encounter is shown in Figure 1.2.3-7. Under worst-case conditions, the link will operate for 2.8 months beyond encounter, while for nominal conditions the link is extended to about 3.3 months (290 x 10^6 km).

Figure 1.2.3-8 shows the Mode 5 medium gain antenna performance for the latest encounter range of 260 x 10^6 km. For this case, communication is possible to 1 month worst-case and to about 2.4 months nominal (345 x 10^6 km).



E_1 = Encounter on Jan. 18, 1974, 150×10^6 km
E_2 = Encounter on Feb. 28, 210×10^6 km
E_3 = Encounter on Apr. 4, 260×10^6 km
E_1 to E_2 = 1.37 Mo
E_2 to E_3 = 1.37 Mo

Figure 1.2.3-6: MODES 1 AND 5 MEDIUM-GAIN ANTENNA PERFORMANCE

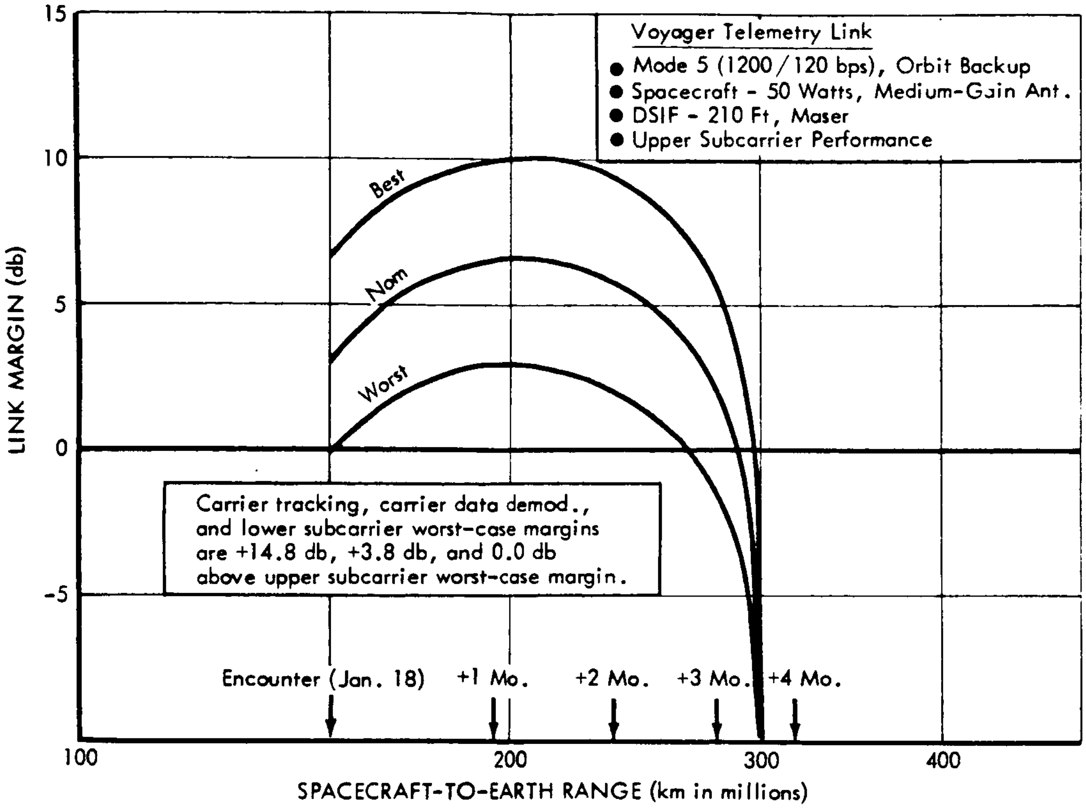


Figure 1.2.3-7: MGA MODE 5 PERFORMANCE — Early Encounter (150×10^6 km)

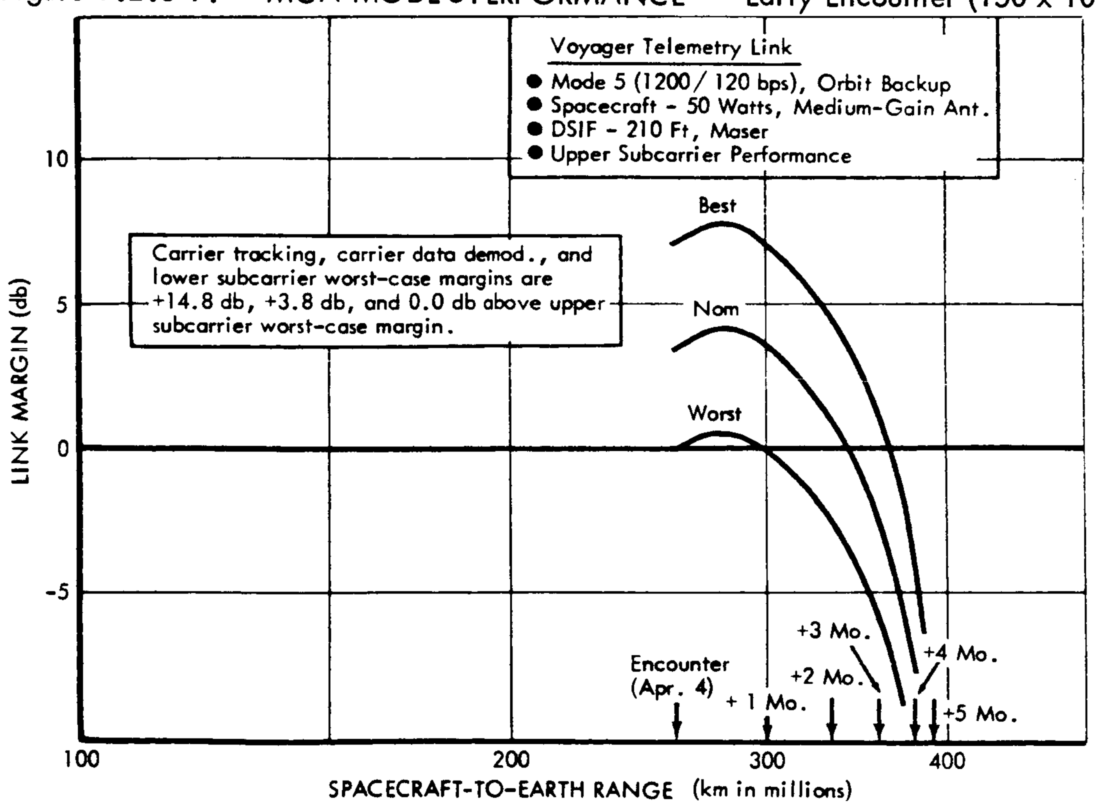


Figure 1.2.3-8: MGA MODE 5 PERFORMANCE — Late Encounter (260×10^6 km)

5) Telemetry Mode 6 -- Mode 6 is a cruise and early-orbit emergency mode providing 8-1/3 bps. The mode has an all-spacecraft-engineering format, and the data rate is chosen to provide both an extended range capability with the low-gain antennas and permit use of the existing Mariner 8-1/3-bps ground demodulators. Table 1.2.3-6 presents the design control table used to evaluate this mode. Note that power is provided for both a data and pseudorandom noise sync subcarrier channel. The peak phase deviations were chosen so that each channel (carrier, data, and sync) thresholds at the same point under worst-case margins. Figure 1.2.3-9 presents the performance graph for Mode 6 and shows the link enters grayout at a range of 160×10^6 km. Under nominal margins, the link operates to 260×10^6 km before blackout and thus covers the spread in possible encounter ranges.

Performance Summary of Primary and Alternate Links -- Analysis of alternate links has been conducted in addition to the primary link design control tables presented in the previous paragraphs. A performance summary chart is given in Table 1.2.3-7 for all the primary and possible alternate links. The table shows the spacecraft and DSIF configurations available in each mission phase and the resultant performance limits. The flexibility of the preferred design provides sufficient additional capability through the use of alternate links in the event of a prime link failure.

1.2.3.2 Earth-to-Spacecraft Command Link

The performance of the Earth-to-spacecraft command link for all possible DSIF and spacecraft equipment configurations is established in this section through a design control table, performance graph, and scaled performance summary. The link provides reception at the spacecraft over the omnidirectional low gain antennas. The spacecraft configuration is fully redundant, with each antenna feeding a corresponding receiver preceded by a low-noise preamp for improved reception capability. The Task D command link differs from that of Task B in that a dual-channel command detector, described in Section 1.2.6.2, has been selected as the recommended approach.

Applicable modulation parameters for the command link are summarized in Table 1.2.3-3. The spacecraft receiving configuration is identical to that of Task B, resulting in the same system noise temperature and tracking performance as was derived in the Task B documentation.

The primary command link configuration consists of a 10-kw transmitter diplexed over the 210-foot DSIF antennas, with spacecraft reception occurring over a dual low gain antenna/preamp/receiver combination. A link analysis for this configuration is performed in Table 1.2.3-8, with the resultant link performance being illustrated in Figure 1.2.3-10. As seen, adequate link margin is available under worst-case conditions to well beyond maximum mission range.

Link performance for all possible DSIF configurations is summarized in Table 1.2.3-9, for both nominal and worst-case conditions. Normal mission operation would involve

Table 1.2.3-6: DESIGN CONTROL TABLE - MODE 6

PROJECT: VOYAGER 1973, TASK D DATE:

CHANNEL: SPACECRAFT-EARTH EMERGENCY TELEMETRY LINK, MODE 6 (8-1/3 bps)

MODE: SPACECRAFT - 50W, LGA; DSIF -210 Ft, MASER

NO.	PARAMETER	VALUE	TOLERANCE	SOURCE
1	Total Transmitter Power, **50W**	+ 47.0 dbm	+1.0, -0.0 db	Fig. 1.2.3-1
2	Transmitting Circuit Loss	- 2.5 db	+0.7, -0.7 db	Fig. 1.2.3-1
3	Transmitting Antenna Gain, **LGA**	+ 1.0 db	+1.0, -0.0 db	Fig. 1.2.3-1
4	Transmitting Antenna Pointing Loss	Included in (3)	---	
5	Space Loss (2295 mc, 200 x 10^6 km)	-265.7 db	---	
6	Polarization Loss	- 0.2 db	+0.0, -0.1 db	Table 1.2.3-2
7	Receiving Antenna Gain, **210 FT.**	+ 61.0 db	+1.0, -1.0 db	Table 1.2.3-2
8	Receiving Antenna Pointing Loss	0.1 db	+0.1, -0.1 db	Table 1.2.3-2
9	Receiving Circuit Loss	Included in (7)	---	
10	Net Circuit Loss	-206.5 db	+2.8, -1.9 db	
11	Total Received Power	-159.5 dbm	+3.8, -1.9 db	
12	Receiver Noise Spectral Density (N/B) (T System = 45, ±10 °k)	-182.1 dbm	+0.9, -1.1 db	Table 1.2.3-2
13	Carrier Modulation Loss	- 4.6 db	+0.9, -1.1 db	Table 1.2.3-3
14	Received Carrier Power	-164.1 dbm	+4.7, -3.0 db	
15	Carrier APC Noise BW ($2B_{LO}$ = 5 Hz)	+ 7.0 db	+0.0, -0.5 db	Table 1.2.3-3
	CARRIER PERFORMANCE — TRACKING (ONE WAY)			
16	Threshold SNR in $2B_{LO}$	+ 9.0 db	---	Table 1.2.3-3
17	Threshold Carrier Power	-166.1 dbm	+0.9, -1.6 db	
18	Performance Margin	+ 2.0 db	+6.3, -3.9 db	
	CARRIER PERFORMANCE — TRACKING (TWO WAY)			
19	Threshold SNR in $2B_{LO}$	+ 9.0 db	---	Table 1.2.3-3
20	Threshold Carrier Power	-166.1 dbm	+0.9, -1.6 db	
21	Performance Margin	+ 2.0 db	+6.3, -3.9 db	
	CARRIER PERFORMANCE — DATA DEMODULATION			
22	Threshold SNR in $2B_{LO}$	+ 9.0 db	---	Table 1.2.3-3
23	Threshold Carrier Power	-166.1 dbm	+0.9, -1.6 db	
24	Performance Margin	+ 2.0 db	+6.3, -3.9 db	
	DATA CHANNEL			
25	Modulation Loss	- 4.6 db	+0.8, -0.8 db	Table 1.2.3-3
26	Received Data Subcarrier Power	-164.1 dbm	+4.6, -2.7 db	
27	Bit Rate (1/T), **8-1/3 bps**	+ 9.2 db	---	Table 1.2.3-3
28	Required ST/N/B	+ 6.8 db	+0.3, -0.3 db	Table 1.2.3-3
29	Threshold Subcarrier Power	-166.1 dbm	+1.2, -1.4 db	
30	Performance Margin	+ 2.0 db	+6.0, -3.9 db	
	SYNC CHANNEL			
31	Modulation Loss	- 10.0 db	+1.4, -1.5 db	Table 1.2.3-3
32	Received Sync Subcarrier Power	-169.5 dbm	+5.2, -3.4 db	
33	Sync APC Noise BW($2B_{LO}$ = 0.5 Hz)	- 3.0 db	+0.4, -0.4 db	Table 1.2.3-3
34	Threshold SNR in $2B_{LO}$	+ 12.5 db	+0.3, -0.3 db	Table 1.2.3-3
35	Threshold Subcarrier Power	-172.6 dbm	+1.6, -1.8 db	
36	Performance Margin	+ 3.1 db	+7.0, -5.0 db	

COMMENTS:

Link will operate above 0-db margin to range of 161 x 10^6 km worst case, 252 x 10^6 km nominal.

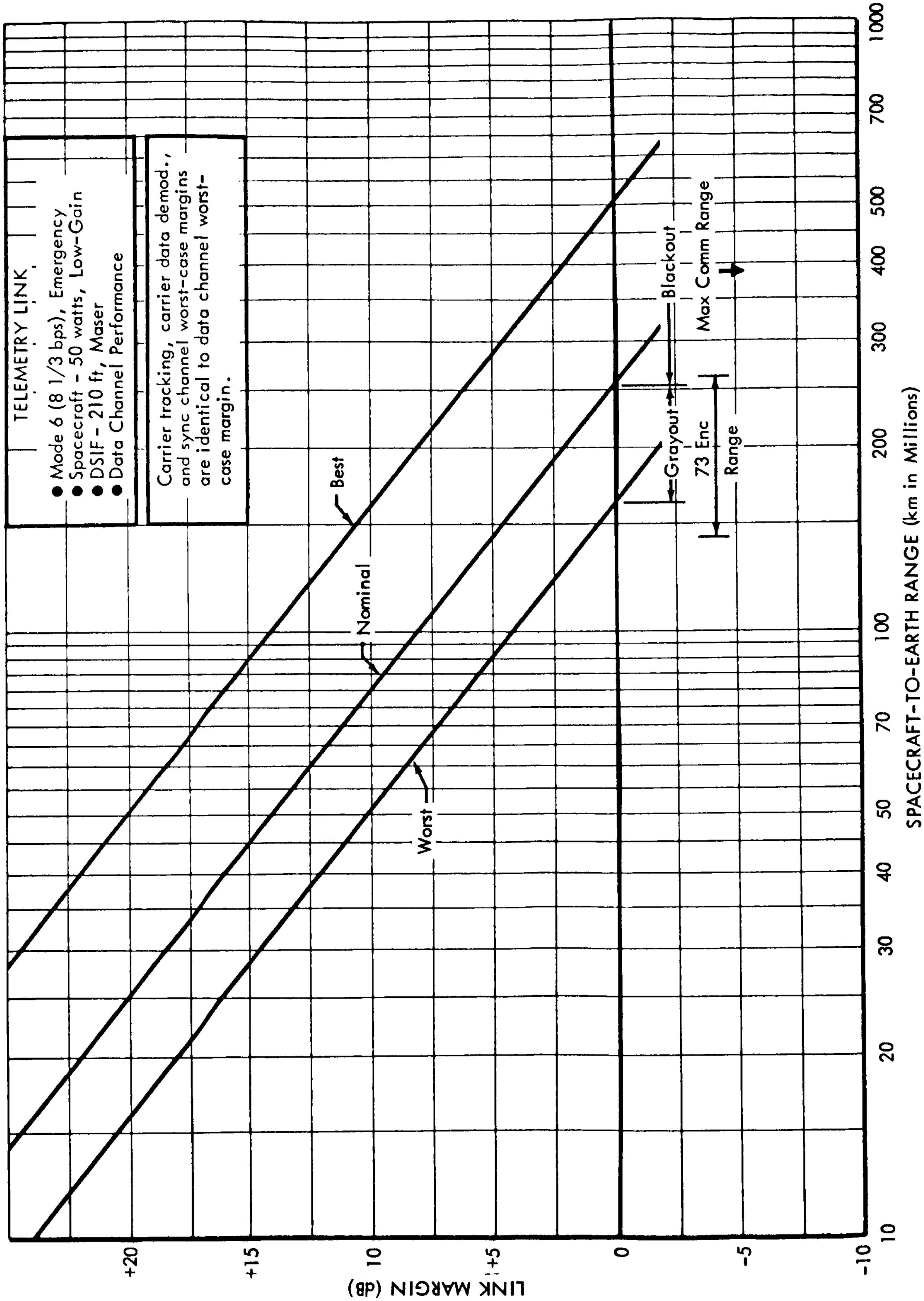


Figure 1.2.3-9: MODE 6 PERFORMANCE

TABLE 1.2.3-7: TELEMETRY LINK PERFORMANCE SUMMARY

MISSION PHASE	MODE (UPPER/LOWER CHANNEL-BPS)	SPACECRAFT		DSIF		DESIGN CONTROL TABLE	RANGE PERFORMANCE LIMITS IN 10^6 KM	
		TRANSMIT POWER	ANTENNA	ANTENNA	$2B_{LO}$, Hz		NOMINAL	WORST CASE
Acquisition and Early Cruise	M1 (75)	1w	LGA	85ft	48	--	1.6	1.16
	M1 (75	1w	LGA	85ft	12	--	3.2	2.32
	M1 (75)	1w	LGA	85ft	5	--	4.96	3.59
	M1 (75)	50w	LGA	85ft	5	--	35.1	25.4
Cruise	M1 (75)	50w	LGA	210ft	5	Table 1.2.3-4	96.5	64.5
	M1 (75)	50w	HGA	210ft	5	--	>>393	>>393
Orbit	M2 (48,000/300)	50w	HGA	210ft	12	Table 1.2.3-5	303	185
	M3 (24,000/300)	50w	HGA	210ft	12	--	> 393	261
	M4 (12,000/300)	50w	HGA	210ft	12	--	> 393	352
Backup	M1 (75)	1w	HGA	210ft	5	--	>>393	>393
	M5 (1200/120)	50w	HGA	210ft	12	--	>>393	>>393
	M5 (1200/120)	50w	MGA	210ft	12	--	345	300
	M1 (75)	50w	MGA	210ft	12	--	375	365
Emergency	M6 (8-1/3)	1w	LGA	85ft	5	--	12.9	8.95
	M6 (8-1/3)	50w	LGA	85ft	5	--	91.5	63.4
	M6 (8-1/3)	50w	LGA	210ft	5	Table 1.2.3-6	252	161
	M6 (8-1/3)	1w	HGA	210ft	5	--	>>393	>>393

Table 1.2.3-8: DESIGN CONTROL TABLE - COMMAND LINK

PROJECT: VOYAGER 1973, TASK D DATE:

CHANNEL: EARTH-SPACECRAFT COMMAND LINK (1 bps)

MODE: DSIF - 10 kw, 210 Ft; SPACECRAFT - LGA, PREAMP

NO.	PARAMETER	VALUE	TOLERANCE	SOURCE
1	Total Transmitter Power, 10 kw	+ 70.0 dbm	+0.5, -0.5 db	Table 1.2.3-2
2	Transmitting Circuit Loss	Included in (3)	---	
3	Transmitting Antenna Gain, 210 Ft.	+ 60.0 db	+0.8, -0.8 db	Table 1.2.3-2
4	Transmitting Antenna Pointing Loss	- 0.1 db	+0.1, -0.1 db	Table 1.2.3-2
5	Space Loss (2115 mc, 200 x 10^6 km)	-265.0 db	---	
6	Polarization Loss	- 0.2 db	+0.1, -0.2 db	Table 1.2.3-2
7	Receiving Antenna Gain, LGA	+ 1.0 db	+1.0, -0.0 db	Fig. 1.2.3-1
8	Receiving Antenna Pointing Loss	Included in (7)	---	
9	Receiving Circuit Loss	- 1.7 db	+0.4, -0.4 db	Fig. 1.2.3-1
10	Net Circuit Loss	-206.0 db	+2.4, -1.5 db	
11	Total Received Power	-136.0 dbm	+2.9, -2.0 db	
12	Receiver Noise Spectral Density (N/B) (T System = 712, + 136, -64°K)	-170.1 dbm	+0.8, -0.4 db	Task B
13	Carrier Modulation Loss	- 1.4 db	+0.3, -0.3 db	Table 1.2.3-3
14	Received Carrier Power	-137.4 dbm	+3.2, -2.3 db	
15	Carrier APC Noise BW ($2B_{LO}$ = 20 Hz)	+ 13.0 db	+0.4, -0.5 db	Table 1.2.3-3

CARRIER PERFORMANCE — TRACKING (ONE WAY)

16	Threshold SNR in $2B_{LO}$	+ 9.0 db	---	Table 1.2.3-3
17	Threshold Carrier Power	-148.1 dbm	+1.2, -0.9 db	
18	Performance Margin	+ 10.7 db	+4.1, -3.5 db	

CARRIER PERFORMANCE — TRACKING (TWO WAY)

19	Threshold SNR in $2B_{LO}$	Not Applicable		
20	Threshold Carrier Power			
21	Performance Margin			

CARRIER PERFORMANCE — DATA DEMODULATION

22	Threshold SNR in $2B_{LO}$	+ 9.0 db	---	Table 1.2.3-3
23	Threshold Carrier Power	-148.1 dbm	+1.2, -0.9 db	
24	Performance Margin	+ 10.7 db	+4.1, -3.5 db	

DATA CHANNEL

25	Modulation Loss	- 11.3 db	+1.0, -1.1 db	Table 1.2.3-3
26	Received Data Subcarrier Power	-147.3 dbm	+3.9, -3.1 db	
27	Bit Rate (1/T), 1 bps	0.0 db	---	Table 1.2.3-3
28	Required ST/N/B	+ 11.7 db	+0.0, -1.0 db	Table 1.2.3-3
29	Threshold Subcarrier Power	-158.4 dbm	+0.8, -1.4 db	
30	Performance Margin	+ 11.1 db	+5.3, -3.9 db	

SYNC CHANNEL

31	Modulation Loss	- 8.4 db	+0.8, -1.0 db	Table 1.2.3-3
32	Received Sync Subcarrier Power	-144.4 dbm	+3.7, -3.0 db	
33	Sync APC Noise BW($2B_{LO}$ =0.4 Hz)	- 4.0 db	+0.0, -1.0 db	Table 1.2.3-3
34	Threshold SNR in $2B_{LO}$	+ 18.7 db	---	Table 1.2.3-3
35	Threshold Subcarrier Power	-155.4 dbm	+0.8, -1.4 db	
36	Performance Margin	+ 11.0 db	+5.1, -3.8 db	

COMMENTS:

Link will operate above 0-db margin to range of 458 x 10^6 km worst case, 710 x 10^6 km nominal.

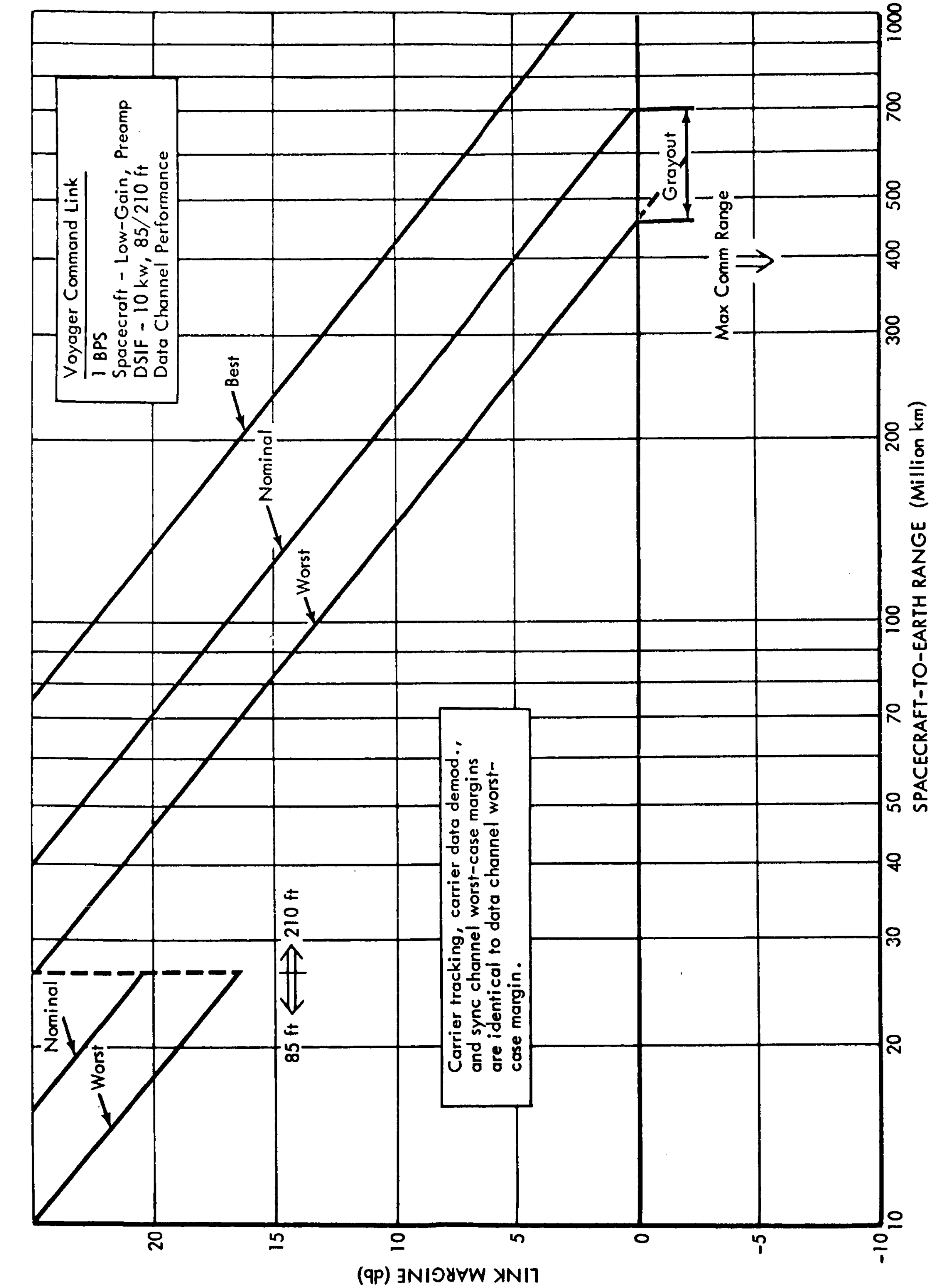


Figure 1.2.3-10: COMMAND PERFORMANCE

CONFIGURATION OR MISSION PHASE	DSIF		DESIGN CONTROL TABLE	PERFORMANCE LIMIT:[a] RANGE, km		LINK MARGIN AT[b] MAX RANGE	
	TRANSMIT POWER	ANTENNA		NOMINAL	WORST CASE	NOMINAL	WORST CASE
Station 72: Ascension	10 kw	30 ft	—	85.5×10^6	54.4×10^6	—	—
Acquisition	10 kw	2 x 2 ft	—	7.03×10^6	4.22×10^6	—	—
Early Cruise	10 kw	85 ft	—	255×10^6	170×10^6	—	—
Maneuver (Carrier Only)	10 kw	85 ft	—	300×10^6	207×10^6	—	—
85 Ft High Power A	25 kw	85 ft	—	>Max Range	269×10^6	+0.2 db	—
85 Ft High Power B	100 kw	85-Ft	—	≫Max Range	>Max Range	+6.2 db	+2.7 db
Midcruise through End of Mission	10 kw	210 ft	Table 1.2.3-8	≫Max Range	>Max Range	+5.1 db	+1.3 db
Maneuver (Carrier Only)	10 kw	210 ft	—	≫Max Range	>Max Range	+6.5 db	+3.0 db
Late Orbit Backup A	100 kw	210 ft	—	≫Max Range	≫Max Range	+15.1 db	+11.3 db
Late Orbit Backup B	400 kw	210 ft	—	≫Max Range	≫Max Range	+21.1 db	+17.3 db

(a) Maximum range (1973): 393×10^6 km

(b) Can accomodate additional link degradation (e.g., antenna mispointing for loss of spacecraft stabilization).

Table 1.2.3-9: COMMAND LINK PERFORMANCE SUMMARY command transmission at 10 kw over the 85-foot DSIF network up to a range of 26×10^6 km. At this range, telemetry performance requires switching to the 210-foot network, which would then be used for command transmission, at 10 kw, through end of mission. Higher power transmission levels would not be required except at extreme range to compensate for antenna pattern degradation in the event of loss of spacecraft stabilization. In cruise and early orbit, the existing margin with 10 kw provides extensive allowance for low gain antenna pointing loss with a degraded spacecraft stabilization.

1.2.3.3 Planetary Ranging Links

Extensive analysis of the planetary ranging link performance was conducted in Task B. This analysis is still valid for Task D with minor changes to reflect the current DSIF parameters. With uplink reception via the low gain antennas, planetary ranging can be achieved for a nominal orbital mission using the 210-foot/10-kw DSIF configuration.

BLANK

1.2.4 Data Storage Subsystem

1.2.4.1 Design Constraints and Requirements
1.2.4.2 Functional Description and Performance Characteristics
1.2.4.3 Physical Characteristics
1.2.4.4 Interface Definition
1.2.4.5 Reliability
1.2.4.6 Trade Study Summary
1.2.4.7 New Technology and Development Items
1.2.4.8 Growth Potential

1.2.4 Data Storage Subsystem

The data storage subsystem (DSS) records spacecraft engineering data, cruise science data, and high rate planetary science data acquired from the science subsystem and the flight capsule for delayed playback and transmission at communication link compatible rates. The selected design consists of eight magnetic tape recorder/reproducers (and interface equipment) with a total data storage capacity of 1.242×10^9 bits. The proposed subsystem weighs an estimated 134 pounds, has a nominal peak power consumption of 54 watts of 2.4-kHz prime power, and occupies about 2 cubic feet.

1.2.4.1 Design Constraints and Requirements

The data storage subsystem is designed to satisfy the following constraints and requirements:

1) Record digital data from the science, telemetry, and radio subsystems.

2) Play back digital data synchronously to the telemetry subsystem.

3) Accept external control of tape speed and record and playback functions for each recorder.

4) Provide a bit error rate of less than 1 in 10^5 bits per recorder.

1.2.4.2 Functional Description and Performance Characteristics

The selected design for the data storage subsystem is shown in Figure 1.2.4-1. The eight recorders of the subsystem are:

1) A flare data recorder of 10^6 bits capacity.

2) A maneuver data recorder of 10^6 bits capacity, with backup capability to record flare data.

3) Two identical recorders, each with 2×10^7 bits capacity, for IR-scanner data, IR-UV spectrometer data, or postlanding capsule data.

4) Four identical high-capacity recorders, each with 3×10^8 bits capacity, for data from the TV sources or from the capsule during descent.

Operating modes for all eight recorders can be controlled both by programmer commands from the computing and sequencing (C&S) subsystem and by real-time commands transmitted from the ground. All recorder commands are routed through the interface and redundancy control unit, which relays the commands to the appropriate recorders on the basis of recorder status as indicated from status inputs to the interface unit from all recorders. Each recorder may provide to the interface unit indications of:

1) Recorder malfunction

2) Unit ready for recording

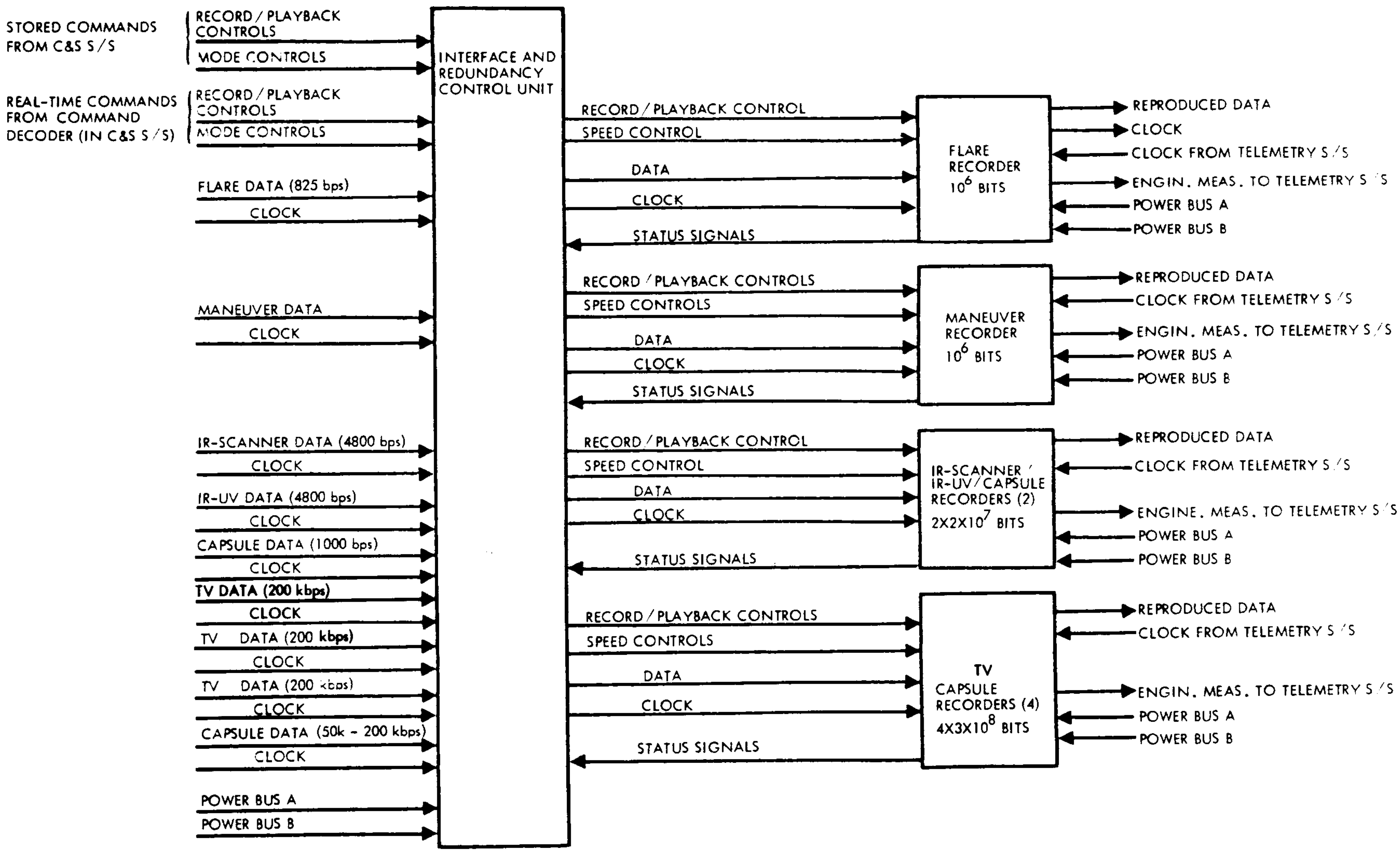


Figure 1.2.4-1: DATA STORAGE SUBSYSTEM FUNCTIONAL DIAGRAM

3) Tape fully loaded

4) Recording in progress

5) Playback in progress.

The recorder malfunction indication is derived in each recorder both from a tape motion-sensing device and detection of playback amplifier signal during recording. In the event either sensor detects recorder failure, a malfunction indication results. Status signals to the interface unit may thus be used for (1) sequencing recorder operation, and (2) detecting malfunctions and switching to backup recorders.

Input data and clock signals to the DSS are similarly routed through the interface unit to be presented to the appropriate recorder on the basis of the status signals received from the recorders. Output data and clock signals from each recorder are routed directly to the telemetry subsystem.

Power buses A and B provide redundant power to the interface unit and each recorder, with isolation provided at each component. If one bus fails, the other is selected. A malfunction within a recorder is not reflected to the power line.

Each recorder provides telemetry measurements directly to the telemetry subsystem for transmission to Earth with other spacecraft engineering data. The seven measurements provided are listed in the interface summary, Section 1.2.4.4.

Performance Characteristics -- Performance characteristics of the recorders are summarized in Table 1.2.4-1. Record mode data rates are selected on the basis of conservative values which might be provided by the science subsystem and the flight capsule. Total science data bit storage capacity is approximately equal to the amount of information that can be transmitted to Earth at 24,000 bps during a 14-hour orbit.

Subsystem Design -- The preferred subsystem design is implemented with magnetic tape recorders, each recorder consisting of a continuous reel-to-reel tape transport and associated electronics. Each recorder is fitted with 1-mil Mylar-backed instrumentation tape. All transports use two motors to cover the various record and playback tape speeds. Speed change of the two-phase hysteresis-synchronous motors is effected by power supply frequency division and attenuation of the voltage supplied. On the basis of a survey of wide motor speed changes, a maximum ratio of 25 to 1 for a single 12-pole motor is used.

The selected transport uses flangeless reels and a peripheral reel-to-reel belt-driven configuration. A single seamless Mylar belt, approximately the same width as the tape, encircles the periphery of the two tape packs and is driven by a differential capstan system. This system is inherently symmetrical, and the tape pack is under positive control of the pressure belt at all times. This provides smooth tape handling during operation, and prevents tape spillage when not operating. The absence of reel flanges permits a substantial reduction in the overall length normally required by a side-to-side, reel-to-reel configuration. Tape edge deformation caused by possible flange riding is thus eliminated.

Table 1.2.4-1: DATA STORAGE SUBSYSTEM PERFORMANCE CHARACTERISTICS

RECORDER	CAPACITY (Bits)	DATA INPUT RATE (bps)	RECORD TAPE SPEED (ips)	PACKING DENSITY (bpi)	PLAYBACK TAPE SPEED (ips)	DATA OUTPUT RATE (bps)	RECORD-TO-REPRODUCE RATIO	TAPE LENGTH (feet)
Flare	10^6	825	3.3	250	0.058 0.092 0.23	14.42 (Avg) 23.07 (Avg) 57.69 (Avg)	57.1:1 Max	340
Maneuver	10^6	75 300 825	0.3 1.2 3.3	250	0.058 0.092 0.23	14.42 (Avg) 23.07 (Avg) 57.69 (Avg)	57.1:1 Max	340
IR-UV/ IR-Scanner/ Capsule (2)	$2 \times 2 \times 10^7$	4800 *1000	2.4 (Two 2-Track Parallel Sections) 0.5	1000	24 12 6 0.6	48,000 24,000 12,000 1,200	1:48 Max	420
TV Capsule (4)	$4 \times 3 \times 10^8$	200,000 *50-200 k	28.6 (Two 7-Track Parallel Sections) 7.15 - 28.6	1000	6.88 3.44 1.72 0.172	48,000 24,000 12,000 1,200	166.7:1 Max	1860

* Estimated for Capsule Data

TV/Capsule Recorders -- Four identical 3×10^8 bit recorders provide for storing high-rate data from the television (photoimagery) sources and from the flight capsule during its high-rate descent mode. Each recorder provides its 3×10^8-bit capacity with 1860 feet, including leader, of a 16-parallel-track, 1-inch tape, using a packaging density of 1000 bits/inch. TV data from the vidicons is recorded simultaneously, each on 7 tracks of the 16-track tape. This parallel-track scheme permits a 7 to 1 reduction in tape speed over a serial record mode, and in addition, permits the recording of the TV sources without prior multiplexing of the signals. Data clock signals for synchronization of each TV signal during playback are recorded on two additional tracks located between the two 7-track sections. As each recorder is filled, an end-of-tape sensor initiates tape rewind, and control inputs at the interface unit transfer the incoming data to another recorder. A playback command to a recorder likewise initiates a rewind if this has not previously occurred, since playback may be desired before a recorder is filled and has reached the end-of-tape sensor. Data is played back, in the same direction as recorded, 7 tracks (plus clock) at a time. After one 7-track section of a recorder is played back, the end-of-tape sensor again initiates rewind for subsequent playback of the remaining 7-track section.

During the flight capsule descent phase when high-rate (50,000 to 200,000 bps) capsule data is received, capsule data has priority over the TV data. When capsule data is presented to the recorders, the TV inputs are overridden and capsule data is recorded on one 7-track section of the recorder in operation. A data clock signal is recorded on an eighth track. One 3×10^8-bit recorder has storage capacity for 12.5 to 50 minutes of capsule data in one 7-track section, depending upon the actual data rate in the 50,000 to 200,000 bps range. Should the recorder in use be filled before capsule transmission is completed, the recorded end-of-tape sensor initiates rewind and activates another recorder for the remaining capsule data.

IR-Scanner/IR-UV/Capsule Recorders -- Two identical 2×10^7-bit recorders provide for storing IR-scanner data and IR-UV data from the science subsystem and, during the capsule postlanding phase, data from the landed capsule. Each recorder provides a 2×10^7-bit capacity with 420 feet of six-parallel-track, 1/2-inch tape, using a packing density of 1000 bits/inch. The IR-scanner and IR-UV data are recorded simultaneously in the same manner as the TV data described above: each source uses two of the six tracks, with a data clock signal occupying a third interior track for each. End-of-tape sensing, rewind, and playback functions are identical to those described above for the TV recorders. When postlanding capsule data, at 1000 bps, is presented to the recorders, it overrides the IR-scanner and IR-UV data just as the high-rate descent capsule data has priority over the TV data.

Flare Recorder -- Flare data, provided by the science subsystem during periods of high Sun activity, is recorded by the flare recorder. A capacity of 10^6 bits is provided using 340 feet of 1/4-inch tape. A packing density of only 250 bits/inch is used to avoid too low a tape speed in playback. An additional track on the tape is used to record the data clock signal.

Maneuver Recorder -- During spacecraft maneuvers, the lower channel digital data stream of the telemetry downlink is provided by the telemetry subsystem to the maneuver recorder for later playback after the maneuver sequence. The maneuver recorder also functions as a backup to the flare recorder, having, in addition to its required record and playback speeds, those speeds required for the flare recorder. The 10^6-bit capacity of the maneuver recorder is provided with the same 340-foot tape complement as is described above for the flare recorder.

1.2.4.3 Physical Characteristics

Data storage subsystem physical characteristics are summarized in Table 1.2.4-2.

Table 1.2.4-2: DATA STORAGE SUBSYSTEM PHYSICAL CHARACTERISTICS

RECORDER	CAPACITY (BITS)	DIMENSIONS (IN.)	VOLUME (CU IN.)	WEIGHT (LB)	PRIME POWER (WATTS) RECORD	PRIME POWER (WATTS) PLAYBACK
Flare	10^6	8 x 6 x 5-1/2	264	12	6	6
Maneuver	10^6	8 x 6 x 5-1/2	264	12	6	6
IR/IR-UV/Capsule No.1	2×10^7	8 x 6 x 5-1/2	264	12	6	6
IR/IR-UV/Capsule No.2	2×10^7	8 x 6 x 5-1/2	264	12	6	6
TV/Capsule No.1	3×10^8	10 x 10 x 5-1/2	550	20	12	6
TV/Capsule No.2	3×10^8	10 x 10 x 5-1/2	550	20	12	6
TV/Capsule No.3	3×10^8	10 x 10 x 5-1/2	550	20	12	6
TV/Capsule No.4	3×10^8	10 x 10 x 5-1/2	550	20	12	6
Interface Unit	--	10 x 2 x 1-1/2	30	6	6	6
TOTALS	1.242×10^9	--	3,286 (~1.9 cu ft)	134.0	54*	

* Peak power consumption

1.2.4.4 Interface Definition

Data storage subsystem interfaces are listed in Table 1.2.4-3.

1.2.4.5 Reliability

Reliability assessments and goal for the data storage subsystem are tabulated below:

Table 1.2.4-3: DATA STORAGE SUBSYSTEM INTERFACES

ITEM	TYPE OF INTERFACE	DESCRIPTION	INTERFACING SUBSYSTEM	INTO DSS	FROM DSS	BOUNDARY DEFINITION
1	Electrical	Prime Power Inputs:				
		a) 50 v rms, 2.4 kHz, Bus A	Power	X		Cable Connector at interface unit and each recorder.
		b) 50 v rms, 2.4 kHz, Bus B	Power	X		
2	Electrical	Input Commands:				
		a) Stored - Record/Playback Controls (2 lines/recorder)	C&S	X		Cable connector at interface unit
		b) Stored - Mode Controls (5 lines)	C&S	X		Cable connector at interface unit
		c) Real-Time - Record/Playback Controls (2 lines/recorder)	C&S	X		Cable connector at interface unit
		d) Real-Time - Mode Controls (5 lines)	C&S	X		Cable connector at interface unit
3	Electrical	Input Data:				
		a) Flare Data (825 bps)	Science	X		Cable connector at interface unit
		b) Clock	Science	X		Cable connector at interface unit
		c) Maneuver Data (75 or 300 bps)	Telemetry	X		Cable connector at interface unit
		d) Clock	Telemetry	X		Cable connector at interface unit
		e) IR-Scanner Data (4800 bps)	Science	X		Cable connector at interface unit
		f) Clock	Science	X		Cable connector at interface unit
		g) IR-UV Data (4800 bps)	Science	X		Cable connector at interface unit
		h) Clock	Science	X		Cable connector at interface unit
		i) Capsule Postlanding Data (1000 bps)	Radio	X		Cable connector at interface unit
		j) Clock	Radio	X		Cable connector at interface unit
		k) TV Data (200 kbps)	Science	X		Cable connector at interface unit
		l) Clock	Science	X		Cable connector at interface unit
		m) TV Data (200 kbps)	Science	X		Cable connector at interface unit
		n) Clock	Science	X		Cable connector at interface unit
		o) TV Data (200 kbps)	Science	X		Cable connector at interface unit
		p) Clock	Science	X		Cable connector at interface unit
		q) Capsule High-Rate Descent Date (50K - 200 kbps)	Radio	X		Cable connector at interface unit
		r) Clock	Radio	X		Cable connector at interface unit
4	Electrical	Input Clock (to each recorder)	Telemetry	X		Cable connector at each recorder
5	Electrical	Reproduced Data:				
		a) Flare Recorder Data	Telemetry		X	Cable connector at each recorder
		b) Clock	Telemetry		X	Cable connector at each recorder
		c) Maneuver Recorder Data	Telemetry		X	Cable connector at each recorder
		d) Clock	Telemetry		X	Cable connector at each recorder
		e) IR-Scanner/IR-UV/Capsule Recorder 1 Data	Telemetry		X	Cable connector at each recorder
		f) Clock	Telemetry		X	Cable connector at each recorder
		g) IR-Scanner/IR-UV/Capsule Recorder 2 Data	Telemetry		X	Cable connector at each recorder
		h) Clock	Telemetry		X	Cable connector at each recorder
		i) TV/Capsule Recorder 1 Data	Telemetry		X	Cable connector at each recorder
		j) TV/Capsule Recorder 2 Data	Telemetry		X	Cable connector at each recorder
		k) TV/Capsule Recorder 3 Data	Telemetry		X	Cable connector at each recorder
		l) TV/Capsule Recorder 4 Data	Telemetry		X	Cable connector at each recorder
6	Electrical	Engineering Measurements (from each recorder):				
		a) End of tape, discrete	Telemetry		X	Cable connector at each recorder
		b) Start of tape, discrete	Telemetry		X	Cable connector at each recorder
		c) Malfunction, discrete	Telemetry		X	Cable connector at each recorder
		d) Tape motion, discrete	Telemetry		X	Cable connector at each recorder
		e) Remaining tape, analog, 0-5v	Telemetry		X	Cable connector at each recorder
		f) Case temperature, analog, 0-5v	Telemetry		X	Cable connector at each recorder
		g) Case pressure, analog, 0-5v	Telemetry		X	Cable connector at each recorder
7	Mechanical	Mounting provision	Structural and Mechanical			Mounting points
8	Thermal	Thermal dissipation	Temperature Control		X	Subsystem Surfaces

DATA STORAGE SUBSYSTEM RELIABILITY

SUBSYSTEM ELEMENTS	Assessed Reliability		Reliability Goal
	Element	Function	
Maneuver Recorder		0.9930	
Flare Data Storage		0.9998	
• Flare Recorder	0.9893		
IR-UV/IR Scanner/Capsule Data		0.9992	
• Recorder No. 1	0.9716		
• Recorder No. 2	0.9716		
TV and Capsule Data		0.9999	
• Recorder No. 1	0.9690		
• Recorder No. 2	0.9690		
• Recorder No. 3	0.9690		
• Recorder No. 4	0.9690		
Total Subsystem (Baseline)	0.992		0.923

Single-Thread Reliability (Assumes all recorders required)	0.818
Baseline configuration, except both IR-UV/IR scanner/capsule recorders required	0.937
Baseline configuration, except both IR-UV/IR scanner/capsule and three of four TV recorders required.	0.932

The data storage subsystem reliability assessment of 0.992 is based on the following:

1) All recorders are off (dormant) except when recording or playing back data.

2) The number of record/playback cycles and the playback time for each cycle is limited by the capability of the telecommunications subsystem to transmit data to Earth.

3) The maneuver recorder is a backup for flare data storage.

4) Any one of the two IR-UV/IR scanner/capsule data recorders is adequate for mission success.

5) Any two of the four TV/capsule data recorders are adequate for mission success.

The above set of success criteria is considered a reasonable basis for reliability assessment purposes. The subsystem does have flexibility for operating in other modes. Several of these alternate modes involving different recorder usage and the corresponding subsystem reliabilities are provided in the table above.

1.2.4.6 Trade Study Summary

During Task D, a trade study was performed between parallel and serial recording techniques. The results are summarized in Figure 1.2.4-2.

1.2.4.7 New Technology and Development Items

The data storage subsystem design is based on existing technology. No new technology or development will be required.

1.2.4.8 Growth Potential

The concept of three types of recorders was selected to provide a system of lowest weight, volume, and power with the required reliability. It is possible to replace these recorders with one recorder type and decrease the total number to six.

The latter concept requires the additional science subsystem function of multiplexing and formatting of telemetry (maneuver) data and flare (825 bps) data to provide a data rate compatible with those of the science recorders (1000 bps to 200 kbps). In addition, all recorders would play back to the upper subcarrier of the telemetry downlinks. These changes allow use of one recorder type with the following characteristics: 16 tracks (two 7-data-track parallel sections and two clock tracks), a maximum packing density of 1000 bits/inch, three record (1200 bps, 4800 bps, and 200 kbps) speeds and four playback speeds (48,000, 24,000, 12,000, and 1200 bps).

Redundancy capability is obviously increased while overall weight is about the same due to deletion of two recorders and replacement of the 10^6- and 10^7-bit units with the common unit (3×10^8 bits capacity).

This one-recorder-type concept has not considered mission profile requirements, but the advantages appear to warrant a more complete study.

TRADE STUDY SUMMARY SHEET	TRADE STUDY TITLE: TAPE RECORDER TRADE STUDY		SELECTION
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	MATRIX OF DESIGN APPROACH		
FUNCTIONAL REQUIREMENTS Input Data: Serial PCM (NRZ) Output Data: Serial PCM (NRZ) **DESIGN REQUIREMENTS** Storage Capacity: 3×10^8 Bits/Recorder Input Data Rate: 1000 - 4,800 bps 50,000 - 200,000 bps Ouput Data Rate: 48,000 bps 24,000 bps 12,000 bps 1,200 bps **SELECTION CRITERIA** 1. Performance 2. Complexity 3. Development status	**PARALLEL DIGITAL RECORDER** DISCUSSION: PRO 1. No track switching 2. Moderate packing densities (< 1000 bpi) 3. Lower tape speed, longer head life 4. Prior space applications CON 1. Additional power 2. Skew problems	**HIGH-DENSITY SERIAL RECORDER** DISCUSSION: PRO 1. No skew problems 2. Lower weight, size, power CON 1. Track switching of record/playback heads 2. Not space-proven 3. Additional development required 4. Higher tape speeds for equivalent input rates	PERFORMANCE Parallel Digital Recorder COMPLEXITY Roughly Equivalent DEVELOPMENT STATUS Parallel Digital Recorder SELECTED APPROACH Parallel Digital Tape Recorder

Figure 1.2.4-2: TAPE RECORDER TRADE STUDY

1.2.5 Telemetry Subsystem

1.2.5.1 Design Constraints and Requirements
1.2.5.2 Functional Description and Performance Characteristics
1.2.5.3 Physical Characteristics
1.2.5.4 Interface Definition
1.2.5.5 Reliability
1.2.5.6 New Technology and Development Items
1.2.5.7 Growth Potential

1.2.5 Telemetry Subsystem

The telemetry subsystem provides acquisition, limited conditioning, and formatting of science and engineering data; generation, modulation, and mixing of subcarriers; and presentation of this data to the radio subsystem for transmission to Earth. The preferred design is an extension of that proposed during Tasks A and B, with the necessary modifications to accommodate the revised spacecraft and capsule design and science data requirements. The design incorporates time and frequency multiplexing, biphase modulation, biorthogonal block coding of science data, and synchronization techniques as selected during Tasks A and B. The proposed design is sized at 900 cubic inches and 34 pounds, and requires 13.9 watts of 2.4-kHz prime power.

1.2.5.1 Design Constraints and Requirements

The telemetry subsystem is designed to satisfy the following constraints and requirements:

1) Provide during interplanetary cruise 2.5×10^6 bits per day minimum of cruise science data.

2) Provide during the orbital operations phase 6.5×10^8 bits per day minimum of science data.

3) Provide an overall bit error rate of less than 5×10^{-3} in the telemetry downlinks.

4) Accept, condition if necessary, and format spacecraft engineering and science data; generate, modulate, and mix baseband subcarriers.

5) Provide a selection of downlink telemetry modes to cover the data rate and data sequencing requirements of the various mission phases.

6) Provide backup, reduced-rate modes for transmission over secondary transmitters and/or antennas.

7) Provide in the telemetry formats proper data identification and synchronization.

1.2.5.2 Functional Description and Performance Characteristics

The design for the telemetry subsystem is shown in the functional block diagram of Figure 1.2.5-1. The subsystem consists of a.c./d.c. power supplies, selected signal conditioners, analog-to-digital (A/D) converters, analog and digital multiplexers, clock countdown circuits, programmers, buffers, synchronization generators, biorthogonal block encoders, biphase modulators, and summing circuits. The design provides efficient acquisition and formatting of data while permitting flexibility to accommodate changing data requirements. This flexibility is attained by acquiring and processing the data from each major source, independent of any other data source, and varying the content of the master data frame to accommodate the mission phase data requirements. The following paragraphs describe the selected telemetry modes and the major functional components of the subsystem.

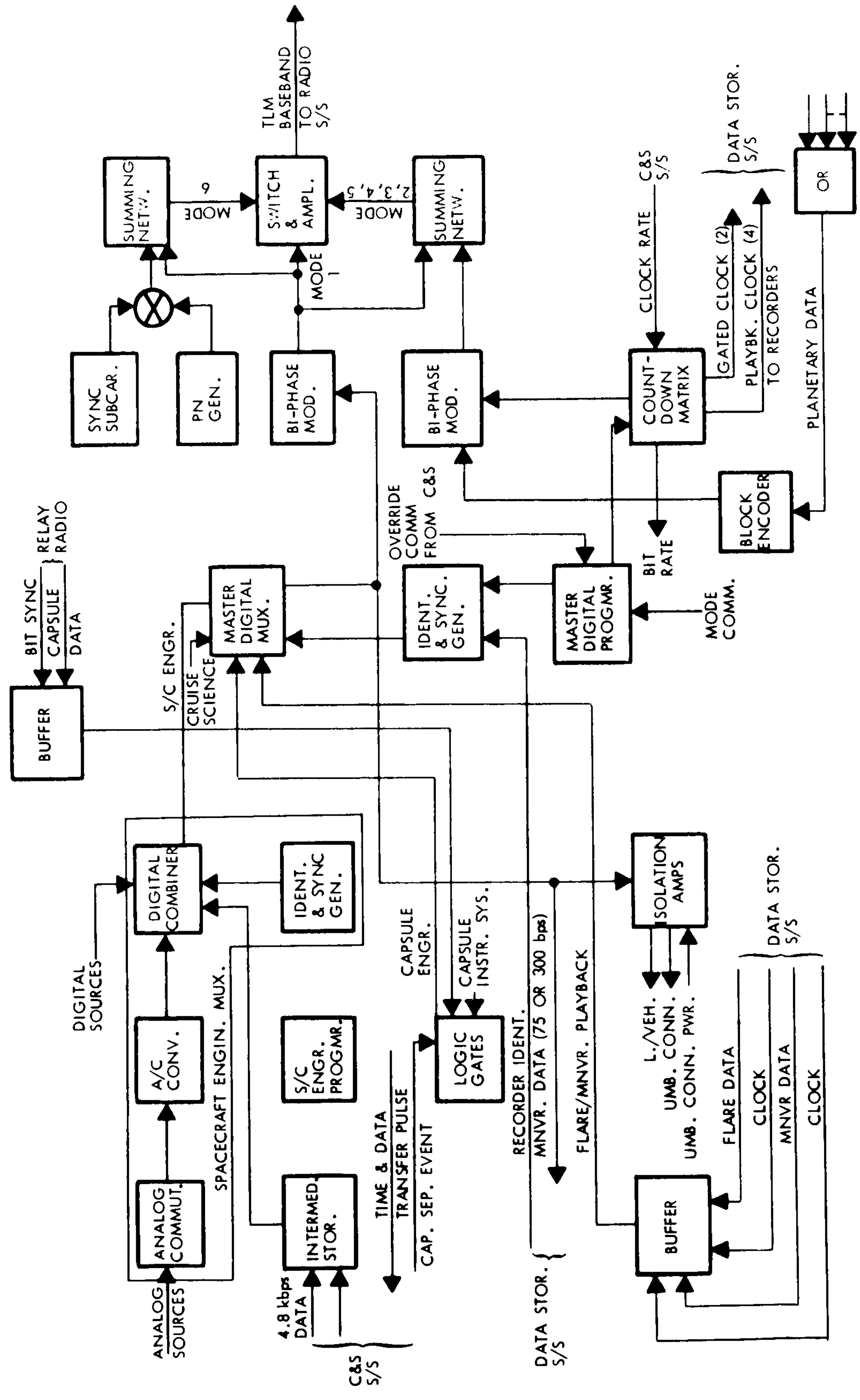


Figure 1.2.5-1: TELEMETRY SUBSYSTEM FUNCTIONAL DIAGRAM

Mode Description -- The telemetry subsystem operates in six basic modes, providing an optimum data content for each mission phase, as follows:

Mode 1 - Launch and Interplanetary Cruise Phase - covers launch to orbit insertion at a data rate of 75 bps.

Mode 2 - Early Orbit Phase - covers orbit insertion, capsule operation (separation, entry, and surface life), and early orbit for early encounter dates or nominal link operation; provides operation to 185×10^6 km at a data rate of 48,300 bps.

Mode 3 - Orbit Phase - covers orbit insertion, capsule operation, and early orbit for nominal encounter dates; provides operation to 261×10^6 km at a data rate of 24,300 bps.

Mode 4 - Late Orbit Phase - covers orbit insertion, capsule operation, and early orbit for late encounter dates; provides operation to 352×10^6 km at a data rate of 12,300 bps.

Mode 5 - Early Orbit Backup Phase - covers early orbit to 300×10^6 km as a backup mode using the medium-gain antenna (MGA); provides data rate of 1320 bps.

Mode 6 - Emergency - provides emergency transmission at 8 1/3 bps over a low gain antenna for ranges up to 160×10^6 km (early encounter date).

All link performance is based on worst-case tolerances.

Mode Parameters -- The mode parameters of the six telemetry modes are summarized in Table 1.2.5-1. A two-channel frequency-multiplexed scheme is adopted, with the lower channel handling spacecraft engineering, cruise science, and capsule engineering data and the upper channel accommodating planetary science data. A lower channel is used in all modes, while the upper channel is added during the orbit phase of the mission, for Modes 2, 3, 4, and 5. Upper channel data is 16,5 biorthogonally coded, providing about a 2-db improvement in link performance, with 16 code bits being transmitted for every five data bits.

Mode 1, used through launch and cruise up to Mars encounter, uses only the lower channel. The 1200-Hz subcarrier handles a bit rate of 75 bps. Modes 2, 3, and 4 are used during the orbit phase of the mission. These modes incorporate a 300-bps lower channel and 48,000, 24,000, or 12,000 data bps (153,600, 76,800, or 38,400 transmitted rate) upper channel. This flexibility permits a mode selection to accommodate available link margin, which will be a function of link tolerances and communication range at encounter and thereafter. The upper channels of Modes 2, 3, and 4 all have the same 614.4-kHz subcarrier frequency, while the lower subcarrier frequency in each case is chosen to provide unambiguous block sync for the upper channel decoder.

Backup capability is provided with Modes 5 and 6. Mode 5 is for use at ranges over 352 million km or with the medium gain antenna during early and midorbit. The lower channel provides 120 bps with a 1200-Hz subcarrier frequency, while the upper channel handles a data bit rate of 1200 bps (3840 bps transmitted bit rate) on a 61.44-kHz subcarrier. The detected lower channel bit rate provides block sync for the upper channel decoder. Mode 6 provides emergency capability using the low gain antennas. In Mode 6, a two-channel transmission scheme is used to provide 8 1/3 bps of spacecraft engineering data. The second channel incorporates a pseudorandom-noise (PN) code for bit and word synchronization.

Major Components -- The major functional components of the subsystem are countdown circuitry for timing signals, the master digital programmer, the master digital multiplexer, the spacecraft engineering multiplexer, including A/D converters, block encoders, and biphase modulators.

Countdown Matrix -- All timing signals, including bit rates and subcarrier frequencies, are obtained from the countdown matrix, Figure 1.2.5-2, which is synchronized with the 3.072-mHz master clock in the computing and sequencing (C&S) subsystem. An independent crystal oscillator clock is also provided in the subsystem for backup should the primary computer and sequencer clock fail.

Table 1.2.5-1: TELEMETRY MODE PARAMETERS

TLM Mode	Lower Channel		Modulation	Upper Channel			Mission Phase(s)
	Bit Rate (bps)	Subc Freq (Hz)	Lower Channel / Upper Channel	Bit Rate (bps)		Subc Freq (kHz)	
				Data	Xmitted		
1	75	1,200	Coherent PSK				Launch, Cruise
2	300	4,800	Coherent PSK / Coded Coherent PSK	48,000	153,600	614.4	Encounter, Early Orbit
3	300	2,400	Same as Mode 2	24,000	76,800	614.4	Orbit
4	300	1,200	Same as Mode 2	12,000	38,400	614.4	Late Orbit
5	120	1,200	Same as Mode 2	1,200	3,840	61.44	Late Orbit, Orbital Backup via MGA
6	8-1/3	75/150	Two-Channel Coherent PSK	---	---	---	Emergency

Basic Clock Frequency = 3.072 MHz

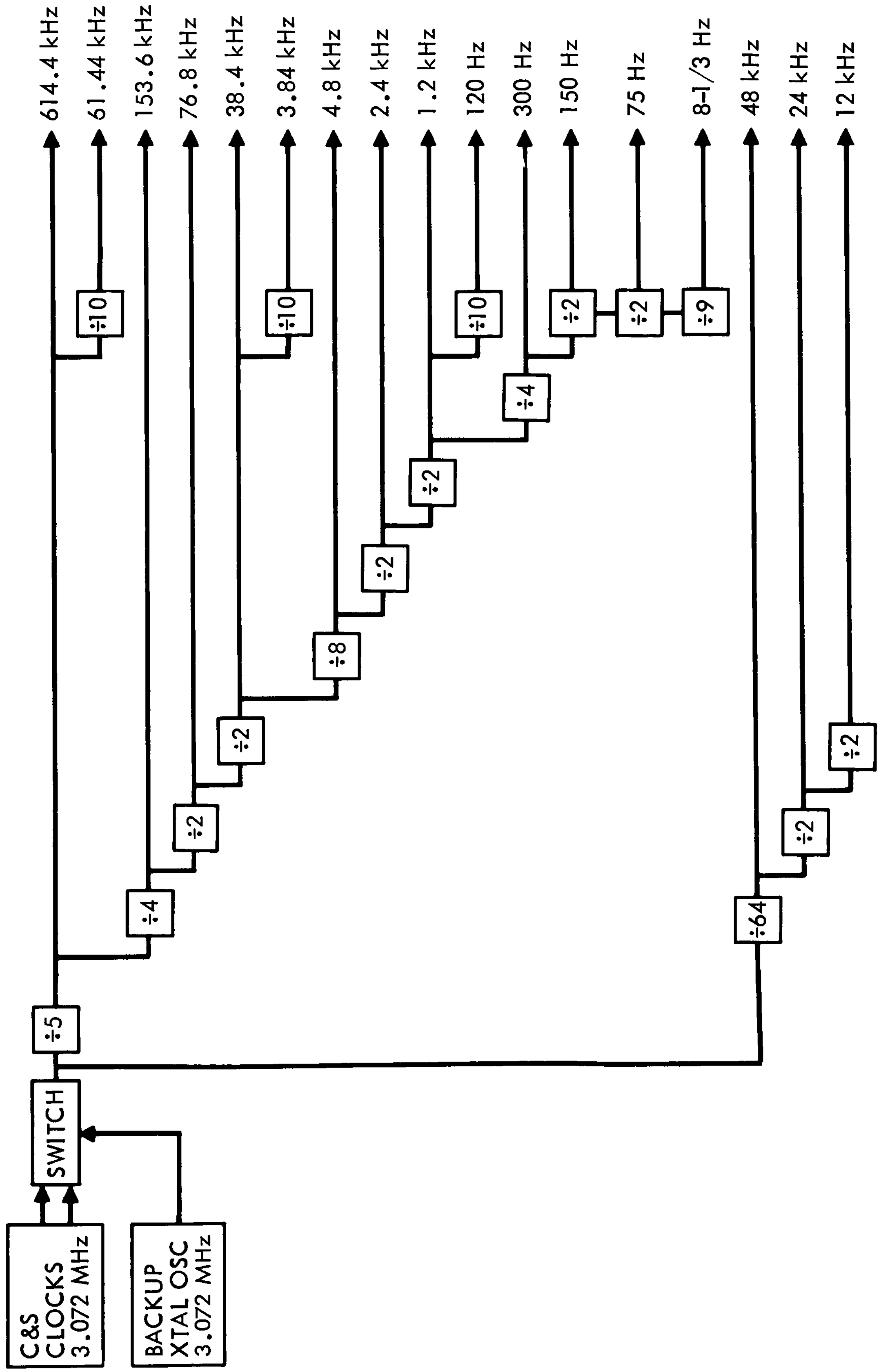


Figure 1.2.5-2: COUNTDOWN MATRIX

Master Digital Programmer -- The master digital programmer for the lower channel contains a basic matrix which is 26 by 5, permitting the programming of a 26-word (7 bits per word) lower-channel telemetry frame from five data sources: (1) frame sync and identification data (I.D.), (2) spacecraft engineering data, (3) capsule engineering data, (4) cruise science data, and (5) recorder playback data. Any of the five data sources could be given any portion of the 26-word frame by adjustment of the matrix. The specific matrix selected for the preferred design permits three telemetry frame formats to be used, depending upon data-type requirements for each telemetry mode. The three formats are shown in Figure 1.2.5-3. All frames incorporate three words for synchronization and identification, followed by 23 words of spacecraft engineering, capsule engineering, cruise science, and/or recorder playback data. As seen, Modes 1 and 5 have the same format, as do Modes 2, 3, and 4. Mode 6 contains only spacecraft engineering data in addition to frame sync and identification. A breakdown of each data type in bits per second for each mode is shown in Table 1.2.5-2.

Table 1.2.5-2: TELEMETRY MODES DATA TYPE BREAKDOWN (bps)

Data Type \ Mode	1	2	3	4	5	6
Lower Channel:						
Spacecraft Eng.	11.5	46.1	46.1	46.1	18.5	7.37
Capsule Eng.	5.8	115.5	115.5	115.5	9.2	---
Cruise Science	34.6	46.1	46.1	46.1	55.4	---
Tape Recorder Playback	14.4	57.7	57.7	57.7	23.1	---
Sync & Frame I.D.	8.7	34.6	34.6	34.6	13.8	0.96
Total	75	300	300	300	120	8 1/3
Upper Channel:						
Planetary Science	---	48,000	24,000	12,000	1,200	---

In each master frame, the frame sync word is the 7-bit Barker code 1110010. The 7-bit frame-count word occurs next, and is incremented by one for each subsequent master frame; ambiguity in frame count is resolved by a time word in the spacecraft engineering major telemetry frame. The third word, for identification data, identifies the mode of telemetry subsystem operation and the source of playback data for both lower and upper channels.

The selection of the three lower channel frame formats is made to provide optimum proportions of each data type for each mission phase. The Mode 1 format accommodates data return requirements for the launch and cruise phase. In orbit, however,

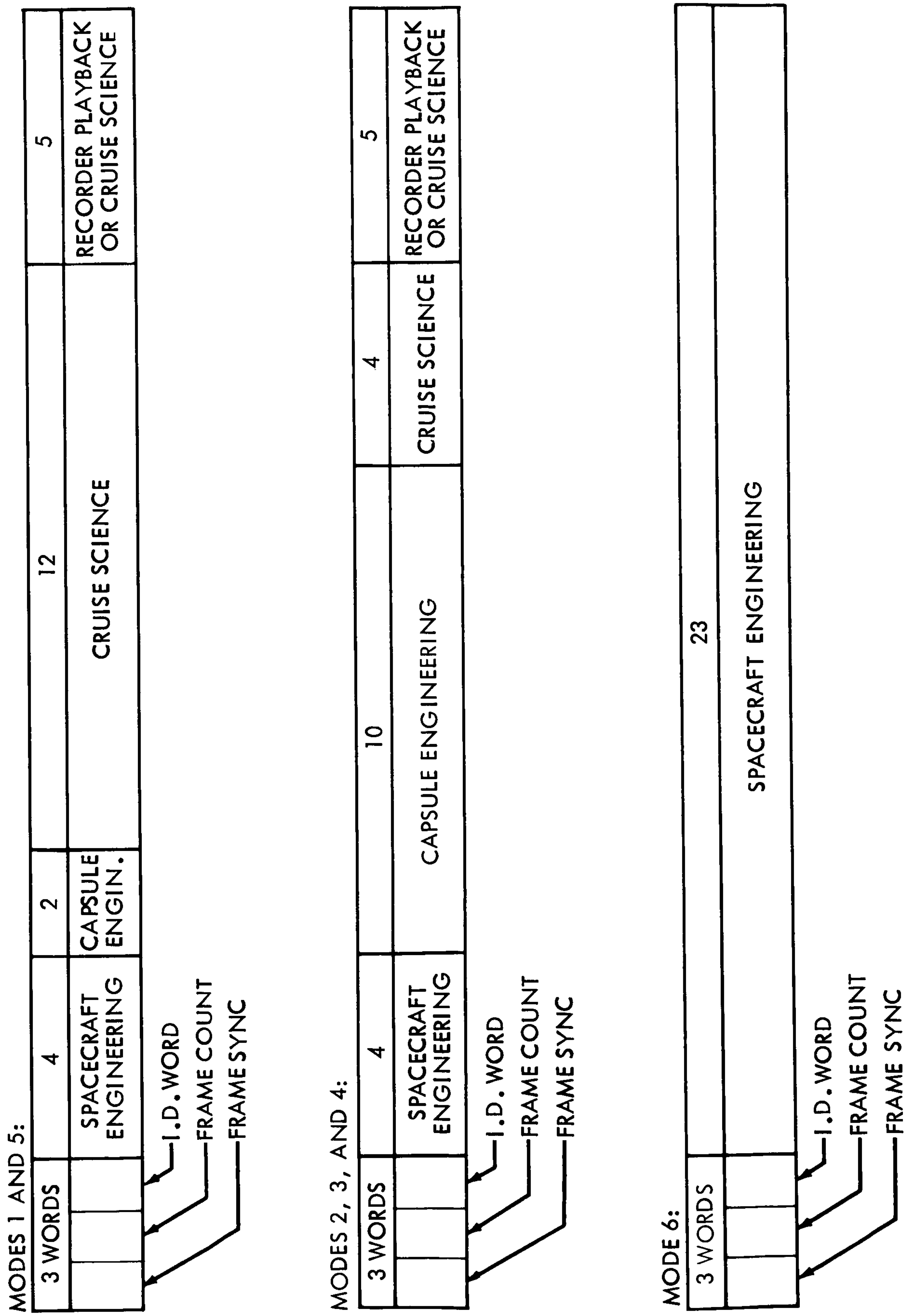


Figure 1.2.5-3: LOWER CHANNEL TELEMETRY FRAME FORMATS

for Modes 2, 3, and 4, the format is changed to provide for the increase in capsule data from capsule system checkout prior to separation and relay of capsule data during descent. Mode 5 reverts to the Mode 1 frame format. Mode 6, for emergency transmission, provides for spacecraft engineering data exclusively.

Master Digital Multiplexer -- The master digital multiplexer is provided all frame construction information and clock signals from the master digital programmer, and in turn constructs the frames of the lower channel data stream from the five data sources that are buffered into it. All timing for the multiplexer is accomplished externally from the countdown matrix. The resultant lower channel bit stream then biphase-modulates the appropriate subcarrier. In Mode 1 operation, the PCM data train and the PCM bit rate are delivered before the biphase modulation process to the planetary vehicle umbilical connector during prelaunch, to the launch vehicle instrumentation unit (IU) during launch, and to the data storage subsystem for recording during spacecraft maneuvers.

Spacecraft Engineering Multiplexer -- The spacecraft engineering multiplexer formats all analog and digital spacecraft engineering data into the major telemetry frame. Spacecraft engineering data consists of those measurements required to determine and verify vehicle and subsystem performance and occurrence of spacecraft events. This data is derived by sequentially sampling analog transducers and digital data sources. Analog data, received at 0 to 5 volts, is digitized into 7 bits, and is multiplexed by a digital combiner with the digital data, received at 0.5 and 4.5 volts, to form an intermittent serial PCM NRZ-L data train. The spacecraft engineering measurements summary is presented in Section 1.1.4. All spacecraft engineering data is then time-multiplexed into a 102-word (714 bits) major telemetry frame. This major telemetry frame is then sampled for as many words at a time as are needed for the lower channel master frame. The initial 15 bits of the major telemetry frame are taken up by the sync word 111101011001000, followed by an 11-bit time word derived from the 11 least significant elements of the computer and sequencer data register. The next two bits identify the spacecraft. The remaining 686 bits of the spacecraft engineering major frame accommodate analog and digital measurements commutated in ratios of 20:1, 10:1, 2:1, not commutated, or 1:6, depending upon the importance of each measurement.

Capsule Engineering Data -- Capsule engineering data provide status information for the capsule both during cruise and after separation until capsule landing on the Martian surface. During cruise, the spacecraft will supply power and bit rate clock signals to the capsule information system, and in turn will receive a hardlined synchronous PCM data train for insertion in the master frame. An average capsule bit rate of 5.8 bps will be transmitted in the Mode 1 format. Prior to capsule separation, the increased average data rate for checkout (115.5 bps) is accommodated by changing to Mode 2, 3, or 4 and increasing the bit rate and word allocation. After capsule separation, 115.5 bps data and derived bit rate are received from the relay receiver of the radio subsystem. These post-separation capsule data are then buffered and shifted out in blocks of 10 words for incorporation in the master frame. It is proposed that the post-separation data rate be equal to or slightly less than the average capsule checkout rate of 115.5 bps, and that synchronization be maintained by inserting a preselected word into the data train whenever the data slips behind the telemetry rate. This artificial word is then identified and discarded on the ground.

Cruise Science Data -- Cruise science data is derived from the science experiments, as a binary bit stream, digitized and formatted with the necessary synchronization and identification. The telemetry subsystem provides a bit rate clock signal, gated as necessary, and in return receives synchronous PCM data, as needed, for real-time inclusion in the time-multiplexed lower channel. Flare data, recorded during periods of high Sun activity, is available from the data storage subsystem at a constant bit rate for inclusion in the words allocated to tape recorder playback. Stored maneuver and flare data from the data storage subsystem is buffered at the reproduce rate, and then read out as needed in response to the gated bit-rate pulses provided by the master digital programmer.

Planetary Science Data -- The upper channel, used for Modes 2, 3, 4 and 5, accommodates planetary science data: photoimagery data from the television recorders, data from the IR scanner and IR/UV spectrometer recorders, and capsule entry and postlanding data from the capsule recorders. Data from any one source is biorthogonally block-encoded and then biphase-modulated on a subcarrier at 48,000, 24,000, 12,000, or 1200 data bps, for Modes 2, 3, 4 and 5, respectively. The corresponding transmitted (coded) bit rates are 153,600, 76,800, 38,400, and 3,840 bps. Data bit rate clock signals are provided to the data storage subsystem by the telemetry subsystem to synchronize the reproduced data. No synchronization or identity information is added to the upper subcarrier data, since the mode identification word in the lower channel master frame contains the identity of the planetary science data.

Subcarrier Modulation -- For Mode 1 transmission, the lower channel data stream biphase-modulates the 1200-Hz sinusoidal subcarrier and the resultant baseband is presented to the radio subsystem. For transmission in Modes 2, 3, 4, and 5, the lower and upper subcarriers are linearly mixed after the biphase modulation processes, and their composite is presented to the radio subsystem.

Mode 6, used for emergency purposes at a low 8-1/3-bps data rate, incorporates, in addition to the data channel, a separate synchronization channel for providing bit and word synchronization. The sync channel subcarrier is modulated by a 63-bit pseudorandom-noise (PN) code coherently generated at 75 bps. The 63-bit sequence is retransmitted for each 7-bit data word, giving 9 PN bits per data bit. The two channels, data and sync, are linearly mixed for presentation to the radio subsystem.

A high-rate telemetry mode for ground checkout is also provided, using the data subcarrier format of Mode 6 (all spacecraft engineering) at a data rate of 1200 bps. A bit-rate clock (1200 bps) is also provided with the data. An enable command via the umbilical initiates the mode.

1.2.5.3 Physical Characteristics

The telemetry subsystem, including mounting, will weigh 34 pounds and have a volume of 900 cubic inches. Consumption of 2.4-kHz prime power will be 13.9 watts average, 16.9 watts peak. Internal power dissipation will be 12.9 watts average, 15.9 peak.

1.2.5.4 Interface Definition

Interfaces between the telemetry subsystem and other spacecraft subsystems are summarized in Table 1.2.5-3.

1.2.5.5 Reliability

Changes made in the telemetry subsystems had a negligible effect on the subsystem reliability. Therefore, the assessed reliability meets the goal which was established by extrapolating previous analyses and configurations to the new mission profile. Reliability assessments for the subsystem elements are tabulated below:

Subsystem Elements	Reliability Assessment	Reliability Goal
Upper Subcarrier	0.99995	
Lower Subcarrier	0.99505	
Total Subsystem	0.995	0.995

1.2.5.6 New Technology and Development Items

The telemetry subsystem design is based on existing technology. No new technology or development will be required.

1.2.5.7 Growth Potential

The existing telemetry subsystem design can be readily modified to incorporate changes in telemetry modes and increases in the number and type of measurement inputs. The present design has the flexibility to satisfy substantial growth requirements before new technology development is required.

Table 1.2.5-3: TELEMETRY SUBSYSTEM INTERFACES

ITEM NO.	TYPE OF INTERFACE	DESCRIPTION	INTERFACING SUBSYSTEM	INTO TM	FROM TM	BOUNDARY DEFINITION
1	Electrical	Engineering Measurements	All subsystems	X		Cable connector at Telemetry S/S
2	Electrical	Real-Time Cruise Science Data - Formatted Serial Digital Pulse Train at 34.6, 46.1, or 55.4 bps	Science	X		Cable connector at Telemetry S/S
3	Electrical	Gates Bit Rate for above	Science		X	Cable connector at Telemetry S/S
4	Electrical	Cruise Capsule Data - Digital Data at 5.8 or 115.5 bps.	Capsule	X		Cable connector at Telemetry S/S
5	Electrical	Gated Bit Rate for above	Capsule		X	Cable connector at Telemetry S/S
6	Electrical	Playback Clock to each recorder (8 signals)	Data Storage		X	Cable connector at Telemetry S/S
7	Electrical	Flare Recorder Data at 14.42, 23.07, or 57.69 bps	Data Storage	X		Cable connector at Telemetry S/S
8	Electrical	Clock for above	Data Storage	X		Cable connector at Telemetry S/S
9	Electrical	Maneuver Recorder Data at 14.42, 23.07, or 57.69 bps	Data Storage	X		Cable connector at Telemetry S/S
10	Electrical	Clock for above	Data Storage	X		Cable connector at Telemetry S/S
11	Electrical	Upper Subcarrier Data - Digital Data at 48,000, 24,000, 12,000 or 1200 bps	Data Storage	X		Cable connector at Telemetry S/S
12	Electrical	Discrete Identifying Data Source	Data Storage	X		Cable connector at Telemetry S/S
13	Electrical	Discrete Separation Capsule Data - Digital Data at 115.5 bps.	Radio	X		Cable connector at Telemetry S/S
14	Electrical	Derived Capsule Bit Rate for above	Radio	X		Cable connector at Telemetry S/S
15	Electrical	Postlanding Capsule Data - Digital Data at 50k - 200k bps	Radio	X		Cable connector at Telemetry S/S
16	Electrical	Clock for above	Radio	X		Cable connector at Telemetry S/S
17	Electrical	Maneuver Data to Maneuver Recorder	Data Storage		X	Cable connector at Telemetry S/S
18	Electrical	Clock for above	Data Storage		X	Cable connector at Telemetry S/S
19	Electrical	Telemetry Baseband to RF Exciters - Frequency - Multiplexed Coherent PSK/PM Signal	Radio		X	Cable connector at Telemetry S/S
20	Electrical	Primary Power - 2.4kHz Square Wave (Buses B & C)	Power	X		Cable connector at Telemetry S/S
21	Electrical	Mode Commands, Clock Rate, and Data	Computing & Sequencing	X		Cable connector at Telemetry S/S
22	Electrical	Transfer Signals (2 Signals)	Computing & Sequencing		X	Cable connector at Telemetry S/S
23	Structural	Subsystem Mounting	Structural & Mech.			Structure
24	Thermal	Control subsystem temperatures throughout all ground test and mission phases	Temperature Control		X	Mounting Points
25	Electrical	3.072 MHz Master Clock Signals (2 Signals)	Computing & Sequencing	X		Cable connector at Telemetry S/S
26	Electrical	Telemetry Data During Prelaunch	OSE		X	Umbilical connector
27	Electrical	Telemetry Data During Launch	Launch Vehicle		X	Cable connector at Telemetry S/S

1.2.6 Radio Subsystem

1.2.6 Radio Subsystem

The radio subsystem is a major element in the RF processing of telemetry, command, tracking, and ranging signals between the spacecraft and the Deep Space Network as well as telemetry signals from the capsule after separation. The following sections present the design details in terms of requirements, functional description, physical characteristics, and interface definition.

1.2.6.1 Design Requirements and Constraints

1) Provide capability for downlink and uplink communications with the flight spacecraft from prelaunch through end of mission.

2) Receive and track in a phase-lock loop a 2115-mHz phase-modulated RF signal transmitted by DSN stations.

3) Coherently convert the received carrier phase and frequency by a 240/221 ratio, and retransmit as an RF carrier frequency at 2295 mHz.

4) Provide an auxiliary oscillator for generating RF carrier frequency in the absence of a received signal for coherent drive.

5) Detect and demodulate the command signal received from DSN stations, and provide data, sync, and lock signal outputs to the command decoder.

6) Demodulate a received PN range code and ranging clock signals, regenerate the range signal with a correlated code, modulate the RF carrier with the range signal, and retransmit.

7) Phase-modulate the carrier with telemetry baseband, and transmit telemetry data to DSN stations.

8) Receive and demodulate the pre- and postlanding RF signal from capsule, and provide the resulting serial data bit streams to the data storage and telemetry subsystems.

1.2.6.2 Functional Description and Performance Characteristics

The radio subsystem of Figure 1.2.6-1 is a key element in performing command, telemetry, and tracking functions between the spacecraft and the ground DSN. The design proposed for Voyager is a fully redundant, S-band system for communications between the spacecraft and Earth. Relay receiving equipment is also included for reception of capsule data during the pre- and postlanding periods.

The Task D design reflects some minor changes from the Task B radio subsystem. The command detectors are now included with the radio equipment and the TWT output has been raised to 50 watts worst case. The subsystem layout, switching configuration, and redundancy options are the same as in the Task B subsystem.

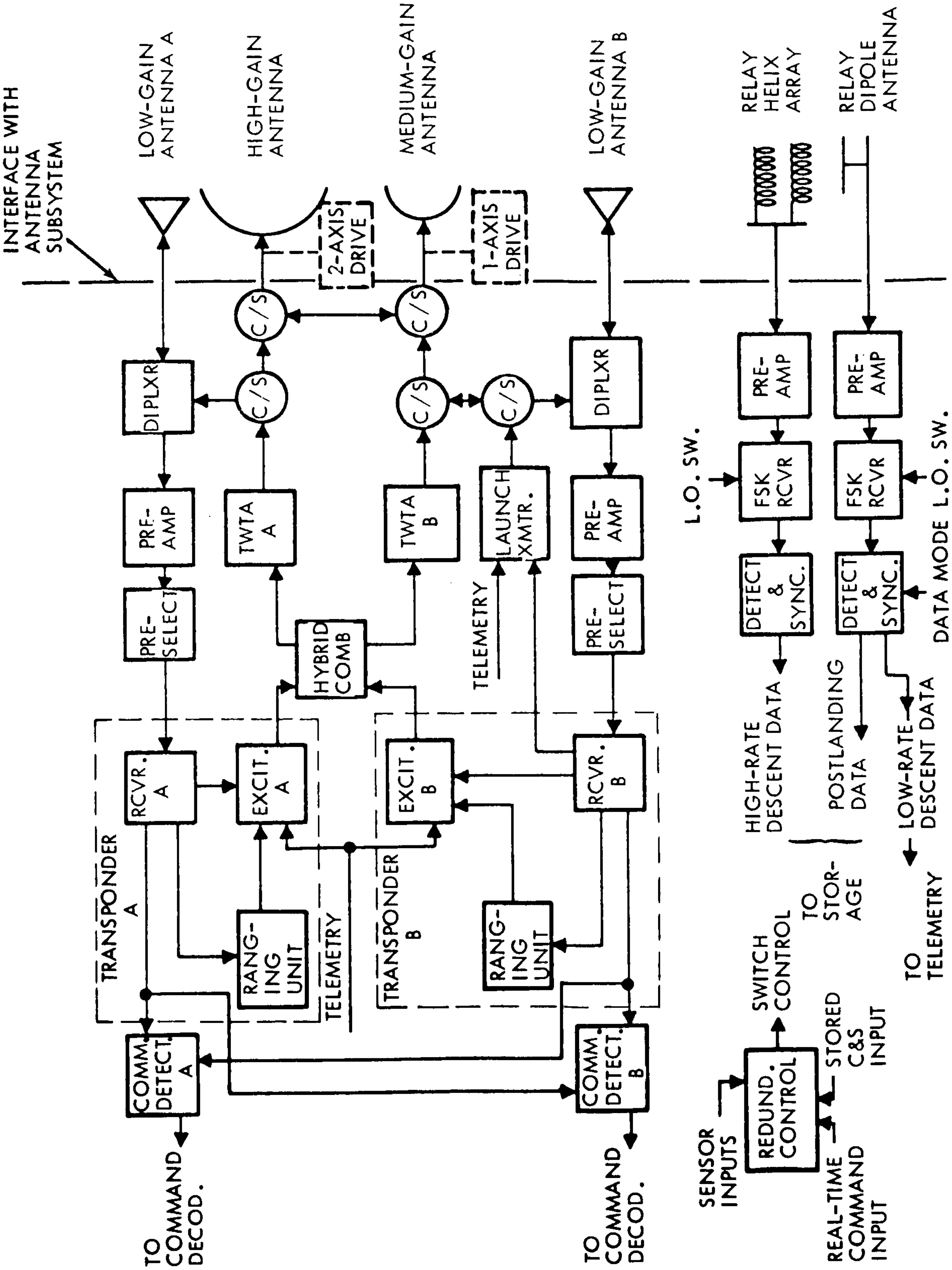


Figure 1.2.6-1: RADIO SUBSYSTEM

Key element trade studies on the command detector, power amplifier, preamplifier, and relay receiving equipment have been completed during the Task D contract. The major technical results of the trade studies are tabulated in the trade study summaries of Section 1.2.6.6.

The following paragraphs discuss the major elements of the subsystem design. Primary emphasis is given to the elements (command detector and relay receivers) that have significantly changed from the Task B design. The specific electrical performance parameters for each component are given in Table 1.2.6-1.

Preamplifier -- A germanium-tunnel-diode preamplifier with a 4.5-db nominal noise figure is used to improve the overall receiving system noise figure. The significant points of the trade study, reaffirming the choice of a tunnel diode, are presented in the Trade Study Summary, Section 1.2.6.6.

Transponder -- The unified S-band transponder consists of the receiver, exciter, and planetary ranging units. The phase-locked receiver demodulates the uplink signals (2115 mHz) and provides output data signals to the command detector and planetary ranging unit as well as the coherent frequency reference signal to the exciter. The ranging unit detects the received PN (pseudorandom-noise) code bits and provides the reconstituted data stream to the exciter for transmission on the downlink to Earth. The exciter accepts and phase-modulates the input baseband telemetry and ranging signals onto the S-band (2295 mHz) carrier frequency. The exciter generates the downlink carrier either from the coherent frequency reference provided by the receiver or, in the absence of the reference frequency, the carrier is derived from the exciter internal crystal oscillator.

The transponder design, except for the ranging unit, is similar to the Mariner '64 and '69 designs. Design improvements resulting from the current Mariner '69 transponder design development will be evaluated and incorporated as appropriate.

Launch Transmitter -- The launch transmitter is similar in design to the transponder exciter unit. The final power amplification and multiplier chain is modified to provide the required 1-watt output power level. The unit is used during launch to conserve prime power and avoid the possibility of ionization breakdown at critical pressure.

Power Amplifier -- The choice of a traveling-wave-tube amplifier (TWTA) as the power amplifier device was reaffirmed during the Task D trade studies. The significant points of the trade study are contained in the Trade Study Summary, Section 1.2.6.6. Extensive development work is being funded by NASA/JPL at both Eimac and Watkins-Johnson on a 50-watt TWT for possible Voyager application. The TWTA operates from the 37 to 108 volts d.c. unregulated power bus.

A 50-watt power level at the tube output under worst-case conditions is specified in the current design. A 1-db allowance is reasonable for tube output variations due to load VSWR, drive level changes, temperature effects, and aging. Thus, the tube output itself is given as +47.0 dbm +1.0 db, -0.0 db.

In addition, a 1.4 db allowance is made in the link analyses of Section 1.2.3 for losses in the tube output filters, isolator, power monitor, and circulator switches.

Table 1.2.6-1: RADIO SUBSYSTEM COMPONENT PERFORMANCE PARAMETERS

PREAMPLIFIER

Center Frequency:	2115 mHz
Bandwidth (3-db):	50 mHz
Noise Figure:	+4.5 db
Gain:	+15 ± 1 db
Gain Compression:	+2 db; -65dbm
Failure Rate:	0.0951%/1000 hours

RECEIVER

RF Frequency:	2115 mHz nominal
Frequency Tracking Range:	±60 kHz
Noise Figure:	+10, +0, -2 db
Threshold:	-151 dbm, +2, -1 db
Dynamic Range:	-70 dbm to threshold
APC Loop Bandwidth:	20 Hz threshold 267 Hz strong signal
AGC Noise Bandwidth:	1.0 Hz threshold 2.0 Hz at -70 dbm
Residual Phase Modulation:	3° peak at -70 dbm
Failure Rate:	1.285%/1000 hours

EXCITER

RF Frequency:	2295 mHz nominal
Power Output:	250 mw
Power Variation:	+1.5 db
Phase Control:	Auxiliary oscillator or Receiver VCO
Aux. OSC Frequency Stability:	$\pm 10^{-7}$ short term $\pm 10^{-6}$ long term
Phase Deviation Capability:	+4 radians
Modulation Frequency Response (+3 db):	1.8 mHz
Spurious RF Outputs:	40 db below unmodulated carrier
Failure Rate:	0.2782%/1000 hours

LAUNCH TRANSMITTER

Same as Exciter Except:	
Power Output:	1.0 watt
Failure Rate:	0.2919%/1000 hours

DIPLEXER

Insertion Loss:	0.5 ±0.1 db receive 0.3 ±0.1 db transmit
Isolation:	100 db xmtr to receiver 40 db receiver to xmtr
Failure Rate:	0.010%/1000 hours

PRESELECTOR

Bandwidth (3-db):	40 mHz
Rejection:	40 db at ±94 mHz 100 db at ± 182 mHz
Failure Rate:	0.007%/1000 hours

POWER AMPLIFIER

Power Output:	50 watts minimum
Saturated Gain:	30 db
Failure Rate:	2.092%/1000 hours

BAND-REJECT FILTER

Center Frequency:	2115 mHz nominal
Rejection:	100 db over ± 10 mHz
Insertion Loss:	0.3 db at 2295 mHz
Failure Rate:	0.010%/1000 hours

BANDPASS FILTER

Insertion Loss:	1.0 db
Bandwidth:	30 mHz
Isolation:	100 db at ± 182 mHz
Failure Rate:	0.010%/1000 hours

RANGING UNIT

Input Frequency:	9.56 mHz
Input Data:	PN code ⊕ clock (received)
Output:	PN code ⊕ clock (regenerated)
Failure Rate:	0.260%/1000 hours

CIRCULATOR SWITCH (3-port)

Insertion Loss:	0.2, +0.2, -0.0 db
Isolation:	20 db minimum
Switching Power:	30 mw
Failure Rate:	0.0211%/1000 hours

CIRCULATOR SWITCH (4-port)

Insertion Loss:	0.3, +0.2, -0 db
Isolation:	20 db minimum per section
Switching Power:	60 mw
Failure Rate:	0.0422%/1000 hours

COMMAND DETECTOR

Bit Rate:	1 bps
Input S/N_0:	+16.5 db·Hz minimum
Output Bit Error Rate:	1×10^{-5}
Synchronization Time:	≤ 12 minutes
Probability of Acquisition:	0.99 at threshold

POWER SUPPLIES

Input Voltage:	50v rms, +3%, -5%; 2.4 kHz ± 0.01%
Regulation:	± 2.0%
Efficiency:	80 - 85%
Failure Rate:	0.085%/1000 hours

POWER SUPPLY (Power Amplifier)

Input Voltage:	37 - 108 vdc
Regulation:	+0.5%
Efficiency:	80% minimum
Failure Rate:	0.085%/1000 hours

Power Supplies -- All active components of the radio subsystem except the TWTA have a.c. power supplies operating from the 2400-Hz, 50 volts rms regulated power buses.

Redundancy Control -- All switching functions in the radio subsystem are accomplished with the redundancy control unit. The unit determines switching requirements based on subsystem sensor inputs, stored commands from the computer and sequencer, or real-time Earth commands from the command path unit.

The redundancy control unit, in conjunction with the RF circulator switching network, can perform any of the following switch functions:

1) Switchover to standby-redundant transponder.

2) Switchover to standby-redundant TWTA.

3) Route TWTA A output to either the low gain A, high gain, or medium gain antennas.

4) Route TWTA B output to either the low gain B, high gain, or medium gain antennas.

5) Route the launch transmitter output to either the low gain B, high gain, or medium gain antennas.

6) Turn on or off both relay receivers, select the operating frequency, and provide for data demodulator switchover after capsule landing.

The implementation of the redundancy control is described in the Task B report.

Command Detector -- The command detector proposed for use in the Voyager Task D design reflects a change from the earlier Task B design in that a two-channel system is used in place of the single-channel system previously recommended (see Section 1.2.6.6). The basic difference in the two designs is that the dual-channel employs two subcarriers for detector operation while the single-channel system uses only one.

The function of the command detector is to detect PCM command data and bit synchronization from the baseband signal obtained from the spacecraft receivers. It serves as an input to the command decoder, providing an input signal consisting of (1) the reconstructed bit stream, (2) bit sync, and (3) a detector "in lock" indication.

The detector, as previously mentioned, is of the two-channel type in which two subcarriers are used to form the baseband signal, one subcarrier for the synchronization signal and the other for the command data. The synchronization signal is a pseudorandom-noise sequence (PN code) which is coherently modulo-2 added to a square-wave subcarrier ($PN \oplus 2 f_s$). The PCM command data is bi-phase modulated on a sinusoidal subcarrier ($\pm \sin w_s t$).

The baseband signal, which is the input to the command detector, is the sum of these two modulated subcarriers. A block diagram of the Mariner '69 two-channel command detector is shown in Figure 1.2.6-2.

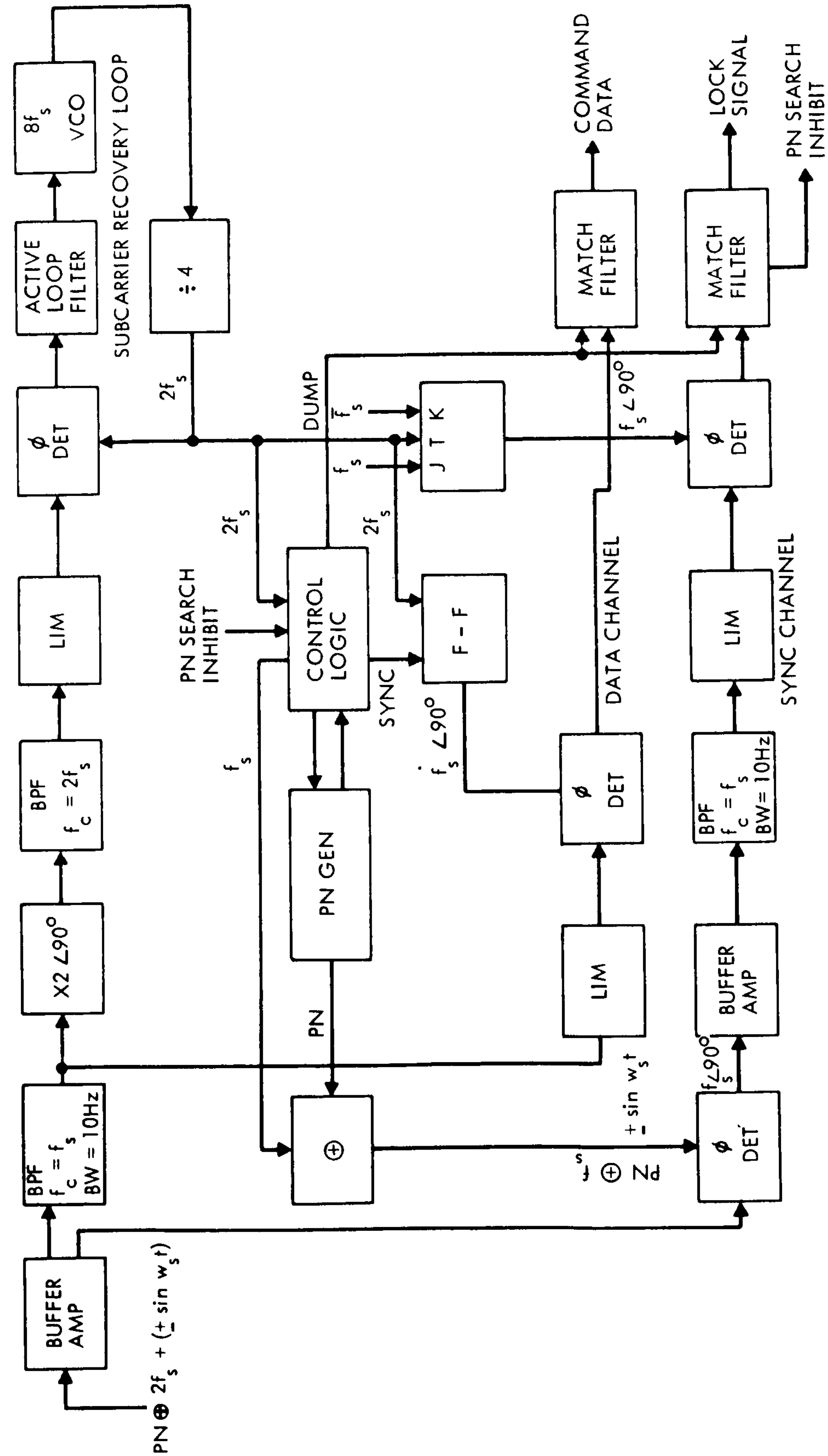


Figure 1.2.6-2: MARINER 1969 DUAL-CHANNEL DETECTOR — Simplified Block Diagram

The detector consists of three basic channels: (1) subcarrier recovery (clock), (2) bit sync (PN correlation), and (3) data.

The subcarrier recovery channel derives the coherent 2 f_s clock frequency for the PN sequence from the data subcarrier, W_s ($2\pi f_s$). This is accomplished by directing the input baseband signal to the narrow-band (10 Hz) bandpass filter, where the synchronization signal is stripped off (See Figure 1.2.6-2). Phase ambiguity is resolved by the x 2 multiplier (full-wave rectifier) which doubles the input frequency and shifts the phase 90 degrees. Unwanted harmonics of the multiplication are filtered in the 2 f_s bandpass filter. The signal is then amplitude limited and directed to the loop phase detector. The phase lock loop coherently locks to the input signal and serves as a source for the coherent 2 f_s clock frequency.

Bit sync is obtained by correlation of the received PN sequence with the locally-generated one. The input baseband signal is directed to the first phase detector (Note 1) in the sync channel. When the ground and airborne PN generators are aligned, the output of the phase detector is $f_s \angle 90$ degrees.

The output of the first phase detector is then amplified, filtered, and amplitude-limited before reaching the second phase detector. Here it is correlated with a locally generated coherent frequency, $f_s \angle 90$ degrees. The output of the second phase detector is detected by a matched filter and an "in lock" signal is generated. The "in lock" indication means that the ground and airborne PN generators are aligned and in time synchronism. A second signal, "PN search inhibit," is also generated when the detector is in lock; this signal inhibits the search mode for the PN generator. When the "PN search inhibit" signal is absent, the PN generator is advanced 1 PN bit per code cycle until correlation (i.e., "in lock") occurs.

The PCM data is detected in the data channel. The baseband input signal is filtered by the narrow-band (10 Hz), bandpass filter, where the synchronization signal PN $\oplus$ 2 f_s is stripped off. The signal is amplitude limited before reaching the command phase detector is then multiplied in the command phase detector by a local, coherent reference frequency.

The output of the phase detector is the reconstructed command data bit stream. The reconstructed data is detected by a matched filter and then passed on to the command decoder.

For the Mariner '69 detector shown in Figure 1.2.6-2, the threshold signal-to-noise-density ratio required at the detector input for a command data error rate of 1 x 10^{-5} is 16.5 db·Hz ($^{+0}_{-1}$ db).

This is a composite signal-to-noise-density ratio, being composed of the sum of the sync channel and data channel signal-to-noise densities.

The power division in the baseband signal is such that the sync channel has twice the power of the data channel. This division results in the following input signal-to-noise densities to the sync and data channels.

$(S/N_o)_{sync}$ 14.74 db·Hz

$(S/N_o)_{data}$ 11.73 db·Hz

The acquisition of the PN code is automatic and occurs when the subcarrier recovery loop achieves lock and the PN generator is advanced until correlation is obtained. The synchronization time is less than or equal to 12 minutes with a probability of acquisition greater than 0.99 at threshold. Other characteristics of the detector are listed below:

Data Bit Rate	1 bps
Prime Power	3 watts
Size	200 cu in.

Relay Receivers -- Significant changes have occurred in the relay receiver design configuration since Task B. The relay receiving system configuration shown in Figure 1.2.6-1 is the result of the trade study summarized in Section 1.2.6.6. A conservative, noncoherent frequency-shift-keyed (FSK) modulation scheme has been assumed for the received capsule signals. A final selection of modulation technique, data rates, and receiver configuration must await a technical interchange and evaluation of system requirements between the capsule and spacecraft contractors and the cognizant NASA centers.

The design selected for Task D is characterized by low-noise preamplifiers preceding the 400-mHz noncoherent-FSK receivers. Each of the two receivers is capable of receiving data from either capsule by selection of the corresponding local oscillator.

One receiver is designed to meet the data requirements associated with the capsule terminal descent TV experiment. This link operates for the final 10 to 15 minutes before landing at data rates of 50,000 to 200,000 bps. The recovered bit stream and sync data from the detector are sent to the data storage subsystem for subsequent playback on the spacecraft-to-Earth link.

The second receiver is designed to handle both the low-rate (~100 bps) capsule data received during the separation-to-landing period as well as the postlanding data received at an assumed 1000 bps. The recovered low-rate descent data and sync are sent from the detector to the telemetry subsystem for real-time transmission to Earth. The postlanding data recovered in the detector is sent to the data storage subsystem for later transmission to Earth.

The high transmission rates available on the spacecraft-to-Earth link offer the possibility of real-time feedthrough of high-data-rate capsule data. This capability could be provided by preempting the upper subcarrier spacecraft science data channel and transmitting capsule data during the period it is being received. Such an option is not currently implemented pending the outcome of the previously mentioned interchange meeting.

1.2.6.3 Physical Characteristics

The physical characteristics for the components of the radio subsystem are tabulated in Table 1.2.6-2.

1.2.6.4 Interface Definition

Table 1.2.6-3 defines the specific subsystem interface characteristics.

Table 1.2.6-3: RADIO SUBSYSTEM INTERFACES

ITEM	TYPE OF INTERFACE	DESCRIPTION	INTERFACING SUBSYSTEM	INPUT TO	INPUT FROM	BOUNDARY DEFINITION
1	Electrical	Telemetry Baseband	Telemetry		X	*Cable connector at radio S/S
2	Electrical	Postseparation Capsule Data - Digital Pulse Train	Telemetry	X		Cable connector at radio S/S
3	Electrical	Derived Capsule Bit Rate - Digital Pulse Train	Telemetry	X		Cable connector at radio S/S
4	Electrical	Radio Subsystem Engineering & Operational Measurements	Telemetry	X		Cable connector at radio S/S
5	Electrical	Backup Operational & Failure Mode Commands (16 Circuits)	Command		X	Cable connector at radio S/S
6	Electrical	Received Uplink Command Baseband	Command	X		Cable connector at radio S/S
7	Electrical	Post Separation Capsule Data - Digital Pulse Train,50-200 kbps	Data Storage	X		Cable connector at radio S/S
8	Electrical	Operational Commands at Specified Times Throughout Mission	Computing & Sequencing		X	Cable connector at radio S/S
9	Electrical	Power Input, 50v rms at 2.4 kHz, Buses B & C	Power		X	Cable connector at radio S/S
10	Electrical	Primary D.C. Power, 37 to 108 vdc (2 signals)	Power		X	Cable connector at radio S/S
11	Electrical	Postlanding Capsule Data - Digital Data Stream at 1000 bps	Data Storage	X		Cable connector at radio S/S
12	Electrical (2)	Phase modulated RF signals received at 2115 mHz (LGA)	Antenna		X	Cable connector at radio S/S
13	Electrical (2)	Phase modulated RF signals transmitted at 2295 mHz (LGA)	Antenna	X		Cable connector at radio S/S
14	Electrical	Phase Modulated RF signals transmitted at 2295 mHz (HGA)	Antenna	X		Cable connector at radio S/S
15	Electrical	Phase Modulated RF signals transmitted at 2295 mHz (MGA)	Antenna	X		Cable connector at radio S/S
16	Electrical	Received RF Capsule signal at 400 mHz, High-Rate (Helix)	Antenna		X	Cable connector at radio S/S
17	Electrical	Received RF Capsule signal at 400 mHz, Low-Rate (Dipole)	Antenna		X	Cable connector at radio S/S
18	Electrical	RF output to Umbilical	Launch Vehicle	X		Cable connector at radio S/S
19	Mechanical	Radio Subsystem Mounting Provisions	Structural & Mechanical	X		Mounting points

*S/S = Subsystem

Table 1.2.6-2: PHYSICAL CHARACTERISTICS - RADIO SUBSYSTEM

COMPONENT	QUANTITY	WEIGHT (Pounds)	POWER (Watts)
Transponder	2		15** ac
Launch Transmitter*	1		
Preamplifier	2	50	
Diplexer	2		
Couplers	2		
Power Amplifiers*	2	16	200 dc
Command Detector	2	13	6 ac
Relay Receiving System	1	14	20 ac
Miscellaneous Attachments		7	--
TOTAL		100	241

*Includes circulator switches.
**24 watts when launch xmtr is used.

1.2.6.5 Reliability

The assessed reliability for the radio subsystem is 0.964. This meets the goal of 0.962 established for this subsystem. Detailed assessments are provided below:

Subsystem Elements	Reliability Assessment	Reliability Goal
Receivers and Ranging Units	0.9845	
Exciters	0.9985	
Power Amplifiers	0.9938	
Launch Transmitter	0.9997	
Descent Relay Radio	0.9975	
Postlanding Relay Radio	0.9971	
Command Detectors	0.9994	
Hybrid Combiner	0.9930	
Total Subsystem	0.964	0.962

1.2.6.6 Trade Study Summary

Trade studies were conducted on the power amplifier, S-band preamplifier, command detector, and the relay receivers during Task D. The results of these studies are summarized in Figures 1.2.6-3 through -6.

1.2.6.7 New Technology and Development Items

No new technology or development is required for the radio subsystem components.

1.2.6.8 Growth Potential

The basic flexibility of the subsystem configuration can accommodate changes both in mission requirements and in interfacing subsystems without any significant re-design. Higher power amplifier levels may be required to support the potential increase in data rate for missions in the late '70's.

TRADE STUDY SUMMARY SHEET

TRADE STUDY TITLE: S-BAND POWER AMPLIFIER

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

POWER AMPLIFIER

Output Power	20 to 100 watts
Gain	20 db or more
Frequency	2295 ± 5 mHz

TRADE

Review of the developmental activities and progress in obtaining an S-band power amplifier device suitable for Voyager application.

TRADE DEVICES

1. Traveling Wave Tube
2. Electrostatically Focused Klystron
3. Amplitron

MATRIX OF DESIGN APPROACH

TRAVELING WAVE TUBE

Characteristics of two traveling wave tubes presently under development contracts are shown below:

EIMAC EM 1249

CHARACTERISTICS

Tube power output	58 watts
Tube efficiency	36%
Gain	30 db
Amplifier efficiency	30.6%*
Prime power	189 watts

*Assuming 85% power supply efficiency

COMMENTS:

1. Cathode current density of 145 ma/cm^2 indicates predicted life of 50,000 hours.

2. Delivery of first engineering model to JPL by July 1967.

W-J - 395

Tube power output	40 to 100 watts
Tube efficiency	43 to 45%
Gain	20 db
Amplifier efficiency	36.5%*
Prime power	274 watts at 100 watts

*Assuming 85% power supply efficiency

COMMENTS:

High output power levels have been achieved.

DISCUSSION:

1. The development programs have made significant progress in the design and testing of a 50 to 100-watt power amplifier.

2. Demonstrated life at lower output levels in space application provides a level of confidence on tube life.

3. Indications are that suitable hardware for the Voyager program can be available in a 36-month program.

ELECTROSTATICALLY FOCUSED KLYSTRON

The characteristics of an electrostatically focused klystron currently being developed under a JPL contract are shown below:

Tube output	20 to 100 watts
Tube efficiency	35% at 20 watts 45% at 100 watts
Gain	30 db min
Amplifier efficiency	29.7% at 20 watts 38.2% at 100 watts
Prime power	147 watts at 50 watts

DISCUSSION:

1. Feasibility model completed and meeting specifications except for some degradation in efficiency.

2. Life tests will be conducted during next phase of development.

3. Design features variable power output.

AMPLITRON

Two-stage 20-watt amplitron is currently being developed for Apollo programs.

DISCUSSION:

Problem areas of inherent low gain and cathode life have not been resolved at this time.

SELECTION

Traveling wave tubes have been designed and tested at 50 - 100 watts under development contracts.

SELECTED APPROACH

Traveling Wave Tube

Figure 1.2.6-3: S-BAND POWER AMPLIFIER TRADE STUDY

TRADE STUDY SUMMARY SHEET

TRADE STUDY TITLE: S-BAND PREAMPLIFIER

MATRIX OF DESIGN APPROACH

SELECTION

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

TRADE

1. Reevaluation current S-band RF preamplifier technology applicable to the Voyager mission.

REQUIRED PREAMPLIFIER CHARACTERISTICS

1. Nominal operating frequency; 2115 mHz $\pm$ 5 mHz
2. Bandwidth 3 db points; 50 mHz
3. Noise Figure; 2 db - 5 db maximum
4. Gain; 15 db $\pm$ 1 db to 20 db $\pm$ 1 db
5. Dynamic Range; 80 db (maximum input level $\leq$ 30 dbm)
6. Phase Characteristics; linear within band limits
7. Failure Mode, Insertion Loss; 7 db maximum

TRADE FACTORS

1. Noise Figure
2. Gain Stability
3. Failure Mode Insertion Loss
4. Development Status

MATRIX OF DESIGN APPROACH

TUNNEL DIODE AMPLIFIER

DISCUSSION:

PRO

1. Low noise figure.
2. Extremely stable gain in 5-port configuration.
3. Low Failure Mode Insertion Loss.
4. Simplicity.

CON

1. No significant improvement in current characteristics are expected. (Minimum development effort required.)

PARAMETRIC AMPLIFIER

DISCUSSION:

PRO

1. Excellent noise figure.
2. Adequate gain.
3. Adequate bandwidth.

CON

1. Temperature compensation to obtain gain stability requires high power consumption.
2. Reliability associated with klystron pump.

COMMENT:

Substantial development effort on solid state pumps to replace the klystron pump previously used by the parametric amplifier is being conducted by industry. A perfected solid state pump may remove most of the objectionable features.

TRANSISTOR AMPLIFIERS

DISCUSSION:

PRO

1. Compact circuit packaging.
2. Can meet minimum preamplifier requirements.

CON

1. Limited gain requires cascading amplifiers.
2. Lack of information on failure mode insertion losses and environmental performance limits.
3. Considerable development effort required.

TRAVELING WAVE TUBE AMPLIFIER

DISCUSSION:

PRO

1. Meets minimum noise figure.

CON

1. The high failure mode insertion loss if the tube fails can be minimized only at the expense of additional complexity and weight.
2. Extreme power supply regulation requirements.

SELECTION

NOISE FIGURE

1. Parametric Amplifier
2. Tunnel Diode Amplifier
3. Transistor Amplifier

GAIN STABILITY

1. Tunnel Diode Amplifier
2. Parametric Amplifier

FAILURE MODE INSERTION LOSS

1. Tunnel Diode Amplifier
2. Parametric Amplifier

DEVELOPMENT STATUS

1. Tunnel Diode Amplifier
2. Transistor Amplifier
3. Parametric Amplifier

SELECTED APPROACH

Tunnel Diode Amplifier

Figure 1.2.6-4: S-BAND PREAMPLIFIER TRADE STUDY

TRADE STUDY SUMMARY SHEET	TRADE STUDY TITLE: COMMAND DETECTOR IMPLEMENTATION	
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	MATRIX OF DESIGN APPROACH	SELECTION

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

The command detector is required for acquisition of the PN synchronization code and detection of the command data.

TRADE

Comparison between the single-channel and two-channel PN synchronization systems.

TRADE CONSIDERATIONS

1. Development
2. Performance

MATRIX OF DESIGN APPROACH

PERFORMANCE COMPARISON - SINGLE CHANNEL VERSUS DUAL CHANNEL

For a bit error rate of 1×10^{-5}

	Single Channel	Mariner '69 Two-Channel
Required Input Signal-to-Noise Density	$18.0^{+0.0}_{-0.0}$ db. Hz	$16.5^{+0.0}_{-1.0}$ db. Hz

Results: The Mariner '69 two-channel system outperforms the single channel system by a factor of $1.5^{+1.0}_{-0.0}$ db

PERFORMANCE IMPROVEMENT - UPDATED MARINER DESIGN

Performance Parameter	Mariner C	Mariner '69
PN Acquisition	Manual	Automatic
Probability of Lock (Threshold)	0.3	>0.99
Input Signal-to-Noise Density for Command Data Error Rate of 1×10^{-5}	18.5 ± 1 db. Hz	16.5^{+0}_{-1} db. Hz
PN Acquisition Time	≈ 30 minutes	≤ 12 minutes

DISCUSSION

1. The two-channel system is presently operating in the breadboard configuration and the design has been accepted for the Mariner '69 Flight Command System.
2. Difficulty in implementation of the single-channel system has resulted in failure to meet anticipated performance level.

SELECTION

DEVELOPMENT

1. Mariner '69 two-channel system

PERFORMANCE

1. Mariner '69 two-channel system

SELECTED APPROACH

Mariner '69 Two-Channel System

Figure 1.2.6-5 : COMMAND DETECTOR IMPLEMENTATION TRADE STUDY

TRADE STUDY SUMMARY SHEET	TRADE STUDY TITLE: RELAY RECEIVER DEFINITION — MATRIX OF DESIGN APPROACH: 1. PCM PSK PM RECEIVER	2. NONCOHERENT FSK RECEIVER	SELECTION
The relay receiver provides the capability for the reception of capsule RF transmission, demodulation, and detection of the capsule scientific and engineering data during capsule descent, landing, and postlanding phases. DESIGN REQUIREMENTS 1. Carrier Frequency — 400 mHz 2. Data input rates, assumed Descent — 100-2,000 bps Descent/Landing — 50,000-200,000 bps Postlanding — 100-2,000 bps 3. Doppler Parameters Maximum Doppler Shift — -4,000 Hz, + 3,630 Hz; -72 Hz/sec, +6.44 Hz/sec ASSUMPTIONS 1. Link Parameters (Typical Descent) Transmitter Power — +43 dbm Transmitting Losses — -3 db Space Loss (400 mHz, 3,240 km) — -154.8 db Total Received Power High data rate — -103.8 dbm Low data rate — -123.8 dbm Receiver Noise Spectral Density — -172.1 dbm 2. Two data link antennas Helical Array — (high data rate) Omnidirectional directed dipole — (low data rate) TRADE CONSIDERATIONS 1. Performance 2. Simplicity 3. Hardware Development	DISCUSSION PRO 1. Requires 4 db less ERP for all data rates; therefore requires less capsule prime power. 2. Minimum doppler effect with proper design. CON 1. Requires search and acquisition of signal. 2. Possibility of false lock or loss of lock.	DISCUSSION PRO 1. Simpler design and lower cost. 2. No acquisition problems; provides continuous link. CON 1. Increased bandwidth to accommodate frequency uncertainty. 2. Requires higher capsule ERP.	PERFORMANCE PM Receiver SIMPLICITY FSK Receiver HARDWARE DEVELOPMENT FSK Receiver SELECTED APPROACH Noncoherent FSK Receiver

Figure 1.2.6-6: RELAY RECEIVER TRADE STUDY

1.2.7 Antenna Subsystem

1.2.7 Antenna Subsystem

The antenna subsystem for the flight spacecraft consists of four S-band antennas, two VHF (400 mHz) antennas, the antenna controllers for the S-band high gain and medium gain antennas, and the structures and transmission lines associated with each antenna.

1.2.7.1 Design Constraints and Requirements

The following design constraints and requirements are satisfied by the antenna subsystem design:

1) Provide for uplink communications with the flight spacecraft from shroud separation through end of mission by a low gain, nonsteerable antenna configuration.

2) Provide a VHF (400 ± 20 mHz) receiving antenna system for the relay link from the capsule from separation through the postlanding period.

3) Provide S-band antennas capable of receiving from Earth in the 2115 ± 5 mHz band and transmitting to Earth in the 2295 ± 5 mHz band.

4) Provide right-hand circularly-polarized antennas for downlink communications with ellipticity not exceeding 0.5 db on axis for the high gain and medium gain antennas.

1.2.7.2. Functional Description and Performance Characteristics

The antenna subsystem, as detailed in Figure 1.2.7-1, is comprised of

1) Two S-band, low gain antenns with fixed omnidirectional toroidal patterns.

2) An S-band, 13.7-foot, two-axis-drive parabolic high gain antenna, with peak gain of 37 db worst case.

3) An S-band, 36-inch, single-axis-drive medium gain antenna, with peak gain of 24 db worst case.

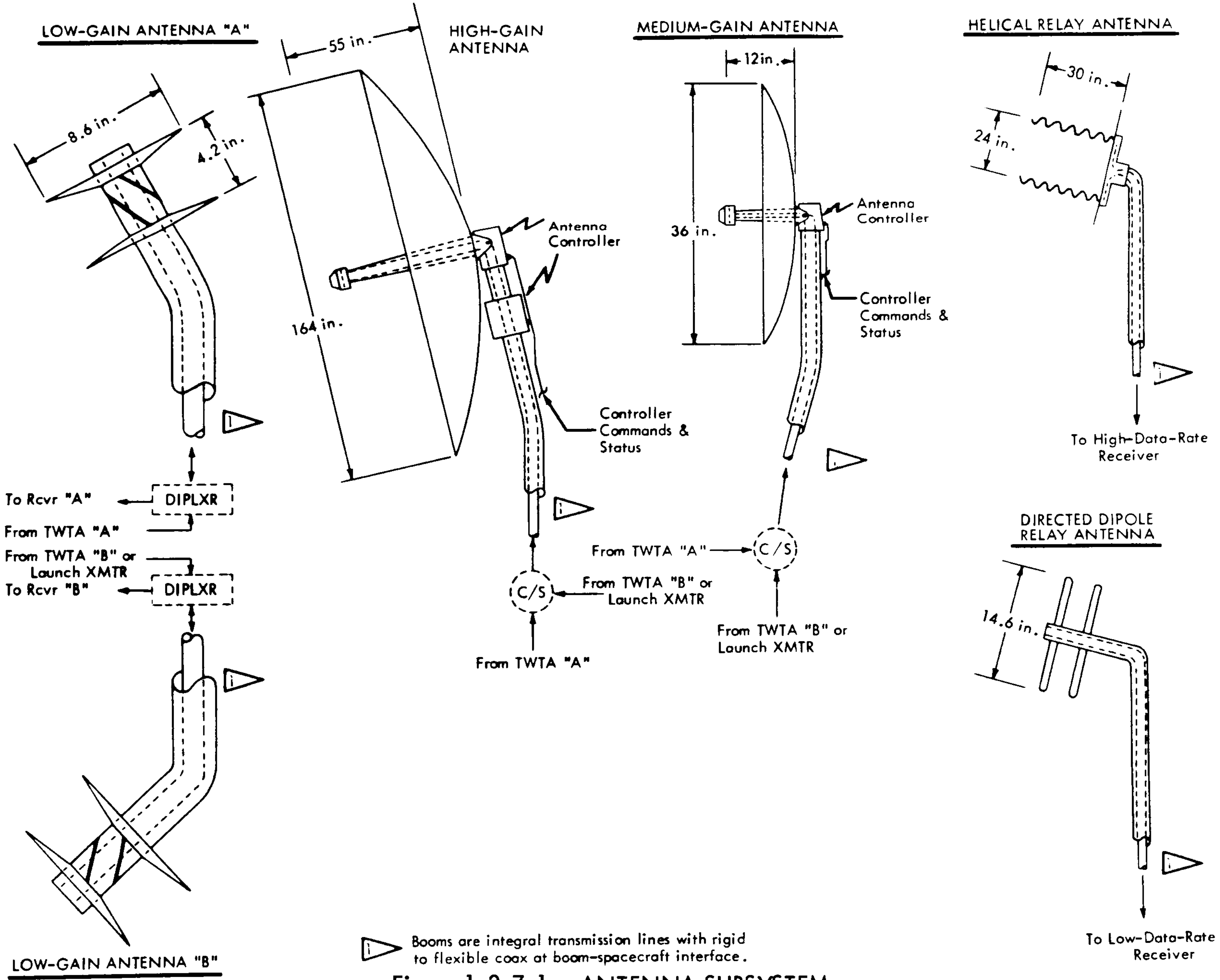
4) A fixed, 400-mHz, directional helical array relay antenna, with a peak gain of 13 db.

5) A fixed, 400-mHz, directed dipole relay antenna.

6) Antenna controllers for the S-band high gain and medium gain antennas.

Low Gain Antennas -- The two low gain antennas (LGA's) are similar to the Lunar Orbiter biconical antenna described in Section 4.1 of the Voyager Task A Report, D2-82709-1. The optimum antenna orientations and boom lengths will be determined by an experimental antenna pattern range study carried out on a scale model of the spacecraft.

High Gain Antenna -- The high gain antenna (HGA) is a paraboloid of revolution having a circular aperture 164 inches (13.7 feet) in diameter, with a gain of



Booms are integral transmission lines with rigid to flexible coax at boom-spacecraft interface.

Figure 1.2.7-1: ANTENNA SUBSYSTEM

37, +1, -0 db and a half-power beamwidth of 2.2 degrees. The focal-length-to-diameter ratio (f/D) will be 0.33, consistent with envelope constraints on the spacecraft prior to deployment.

The feed selected for illuminating the circular paraboloid is a cup-mounted turnstile located at the focus and supported by a central column which is also the outer conductor of a rigid, open, coaxial transmission line. The reflector is a honeycomb-sandwich shell, attached to a tubular support arm by means of a central fitting and star-shaped local reinforcement of the shell. The edge of the shell is stiffened sufficiently to prevent local thermal distortion or buckling.

The shape of the inner surface of the shell is subject to electrical requirements for gain and pointing accuracy. Tolerable errors in the shape of this surface after orbit insertion are expressed as random and systematic deviations from true parabolic contour. The design goal for these errors is 0.14 inch for systematic edge deflections, and 0.32 inch peak-to-peak for random deviations.

The antenna is mounted on a deployed, two-axis steering mechanism comprised of integral coax transmission lines and rotary joints. The steering mechanism is driven by a control system which will maintain the gain of the antenna toward Earth within the -1.9 db-contour off peak gain, corresponding to 0.6 degree per axis in each of two orthogonal axes. The design of the pointing controls is described on page 1-207.

Figure 1.2.7-2 illustrates the range of Earth look angles during late cruise and orbit, beginning at the point where the LGA link performance is at the threshold level and transmission over the HGA is begun. The plot shows the look angles for three cases of the 1973 launch window: (1) earliest, (2) average, and (3) latest date of arrival. The orbit period is shown for 6 months after the latest date of arrival. The required steering angles for the HGA controller mechanism will be 36 degrees for one axis and 40 to 45 degrees on the other. The possible angles for orbit insertion are over a wider range; however, the axes of rotation have been picked to accommodate these angles.

The orientation of the rotation axes are:

1) Principal Axis: Parallel to a line at a cone angle of 90 degrees, clock angle of 185 degrees.

2) Secondary Axis: Parallel to a line at a cone angle of 90 degrees, clock angle of 275 degrees.

Medium Gain Antenna -- The medium gain antenna (MGA) is used as a backup for the HGA. The MGA design is similar to that of the 36-inch Lunar Orbiter high gain antenna. The polarization will be right-hand circular with an axial ratio no greater than 1.59 over the half-power beamwidth. The operating frequency is from 2290 to 2300 mHz. The peak gain of the antenna is 24.0, +0.5, -0 db. The swivel mount is capable of steering in one axis. The ecliptic plane has been selected as the steering plane, based on a desire to optimize coverage during the encounter and orbital period.

Relay Link Antennas -- The VHF (400 mHz) antennas selected for the relay link are (a) an array of two helical antennas for the high-rate mode needed during entry and (b) a directed dipole for the low-rate modes used during descent and postlanding.

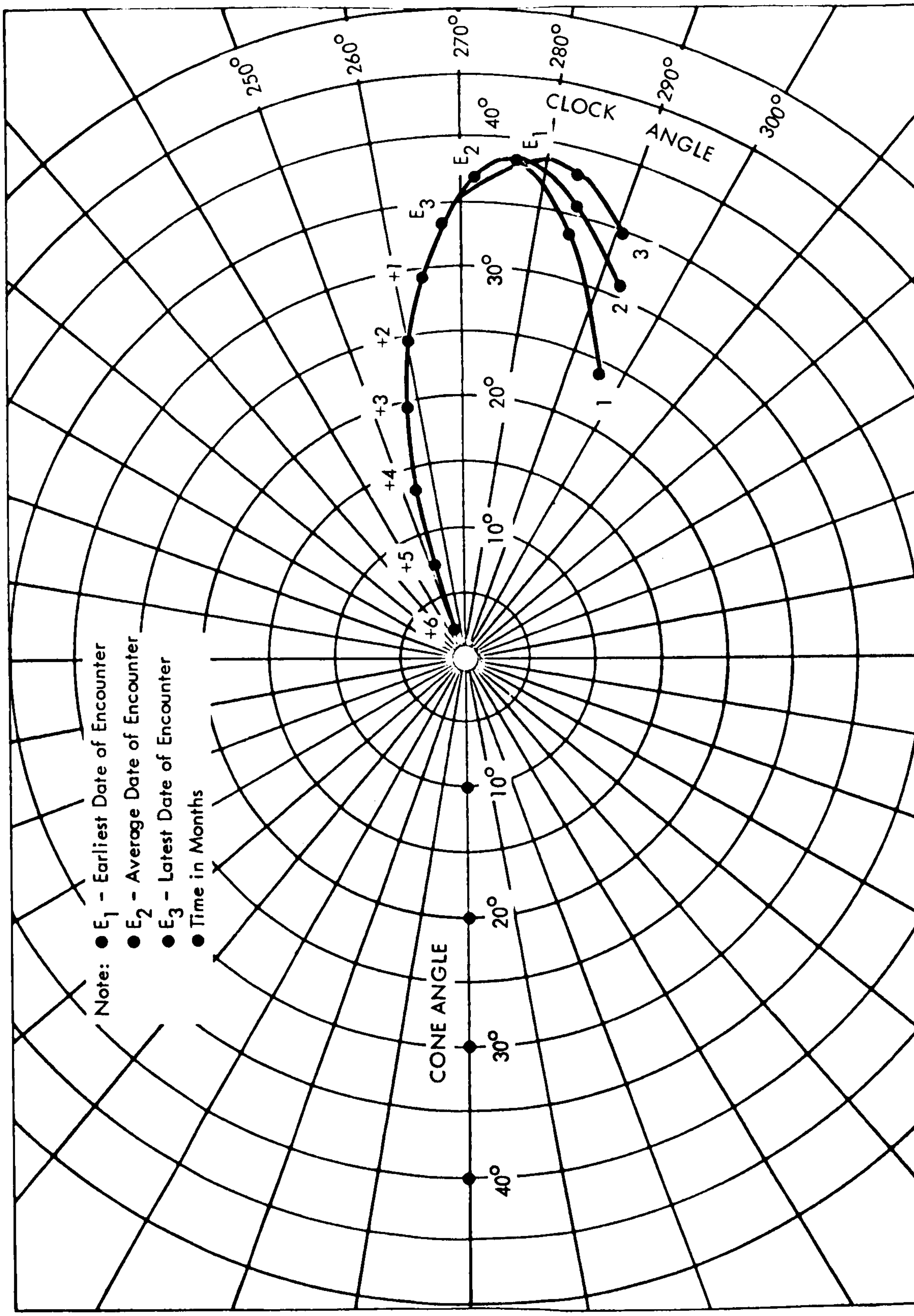


Figure 1.2.7-2: HIGH-GAIN ANTENNA PATTERN COVERAGE OF EARTH CONE AND CLOCK ANGLE - 1973

The helical antenna array (Figure 1.2.7-3) will have a half-power beamwidth of 60 degrees in the orbital plane and 39 degrees out of the orbital plane. This beamwidth is sufficient to provide the coverage required for the various possible trajectories that the capsule could follow. The array of helical antennas will have a peak gain of 13 db.

The directed dipole (Figure 1.2.7-4) will provide the spacecraft with 360 degrees coverage in the plane of the orbit. The gain pattern in the plane of the orbit will be tailored to provide uniform coupling between spacecraft and capsule antennas over the orbital path (Figure 1.2.7-5). It should be noted here that although the orbit precession about Mars is on the order of 1/2 degree per day, the advantage of tailoring the gain pattern to achieve uniform coupling can be realized for several months after separation. The beamwidth of the directed dipole out of the plane of the orbit will be about 100 degrees. The directed dipole will have a peak gain of 5 db, with front-to-back ratio of 14 db. Since the dipole will present linear polarization, the coupling between the circular polarized antenna on the capsule and this antenna will experience at least 3-db polarization loss.

HGA Controller -- The position controller for the HGA is identical for the two axes (this type of control unit is also used for the MGA, science scan platform, and UV spectrometer). A stepping motor and digital encoder are housed inside a harmonic drive gear reduction. The stepping motor and gearing produce antenna motion in increments of 0.2 degree so that antenna drive position can be selected to within ± 0.1 degree. Antenna drive is commanded either by computer and sequencer stored inputs or by real-time command decoder inputs. Detailed design descriptions of the controllers are given in the structural and mechanical subsystem, Section 1.2.10.

Performance Characteristics -- Table 1.2.7-1 presents all performance characteristics of the major elements of the antenna subsystem.

1.2.7.3 Physical Characteristics

Weights of the antenna subsystem components are presented in Table 1.2.7-2. Other physical characteristics are shown in Figures 1.2.7-1, 1.2.7-3, and 1.2.7-4.

1.2.7.4 Interface Definition

Antenna subsystem interfaces are summarized in Table 1.2.7-3.

1.2.7.5 Reliability and Safety

Reliability goal and assessments for the antenna subsystem, for the 1973 mission, are shown in Table 1.2.7-4.

Subsystem reliability studies were based on the backup operational mode using the medium gain antenna in the event that the high gain antenna should fail. This mode yields a subsystem reliability assessment of 0.9980, as reflected by the right column in the table above. In the event that worst-case conditions should require the high gain antenna to transmit all data back to Earth, the subsystem reliability is 0.9835 as reflected by the center column of the table above.

Safety -- Previous studies have indicated that the RF radiation at the near-field boundary of the high gain antenna during integrated system checkout of the power amplifiers may exceed the 0.01 watts/cm^2 allowable maximum limit for personnel

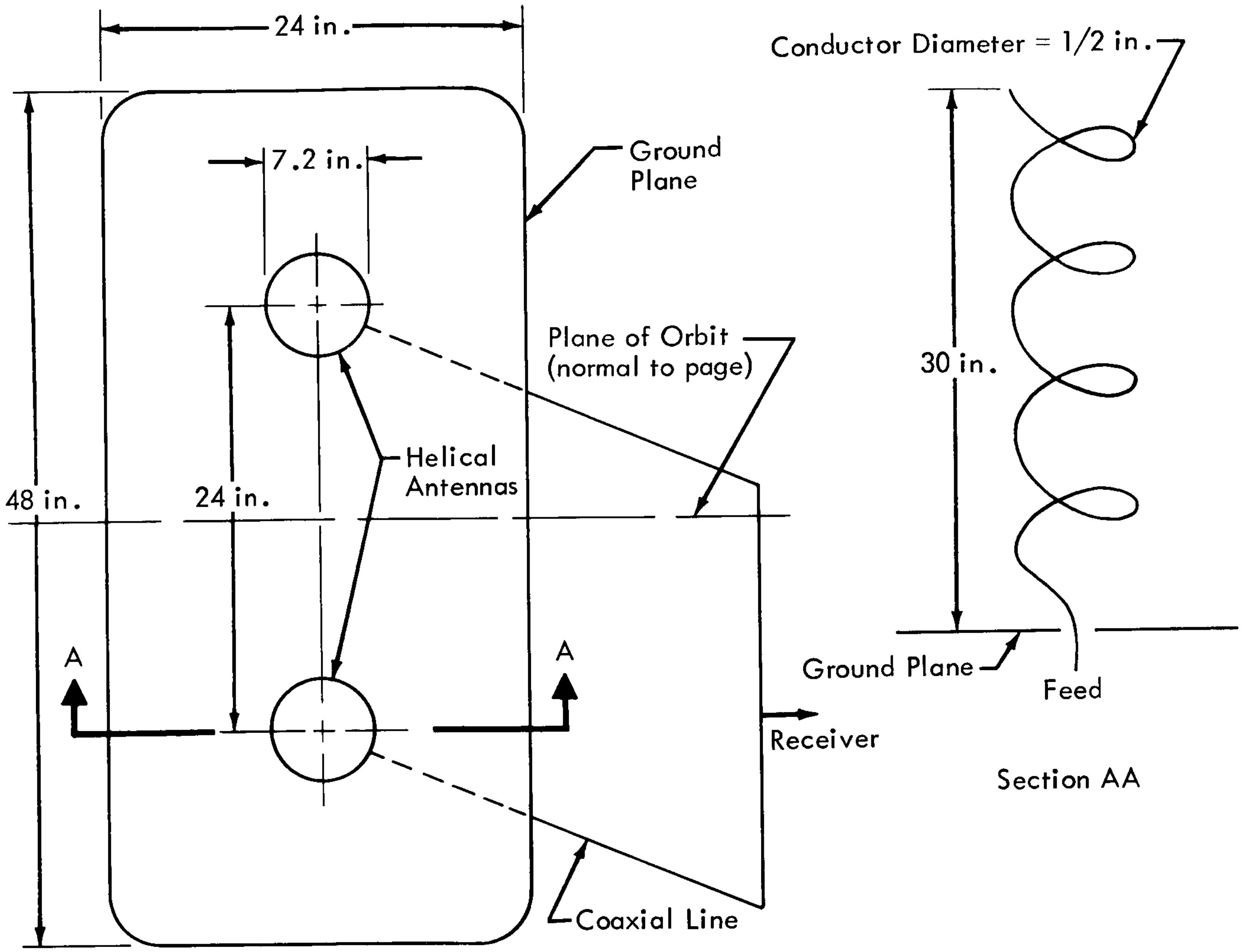


Figure 1.2.7-3: RELAY HELIX ARRAY

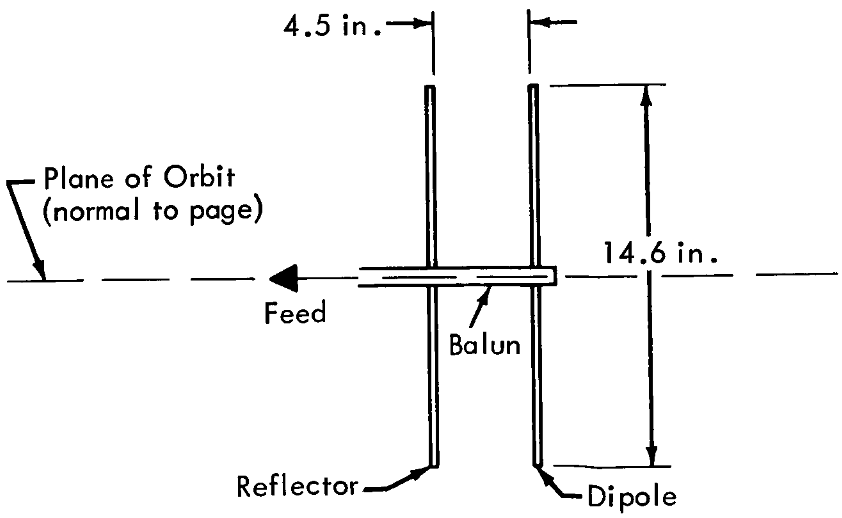
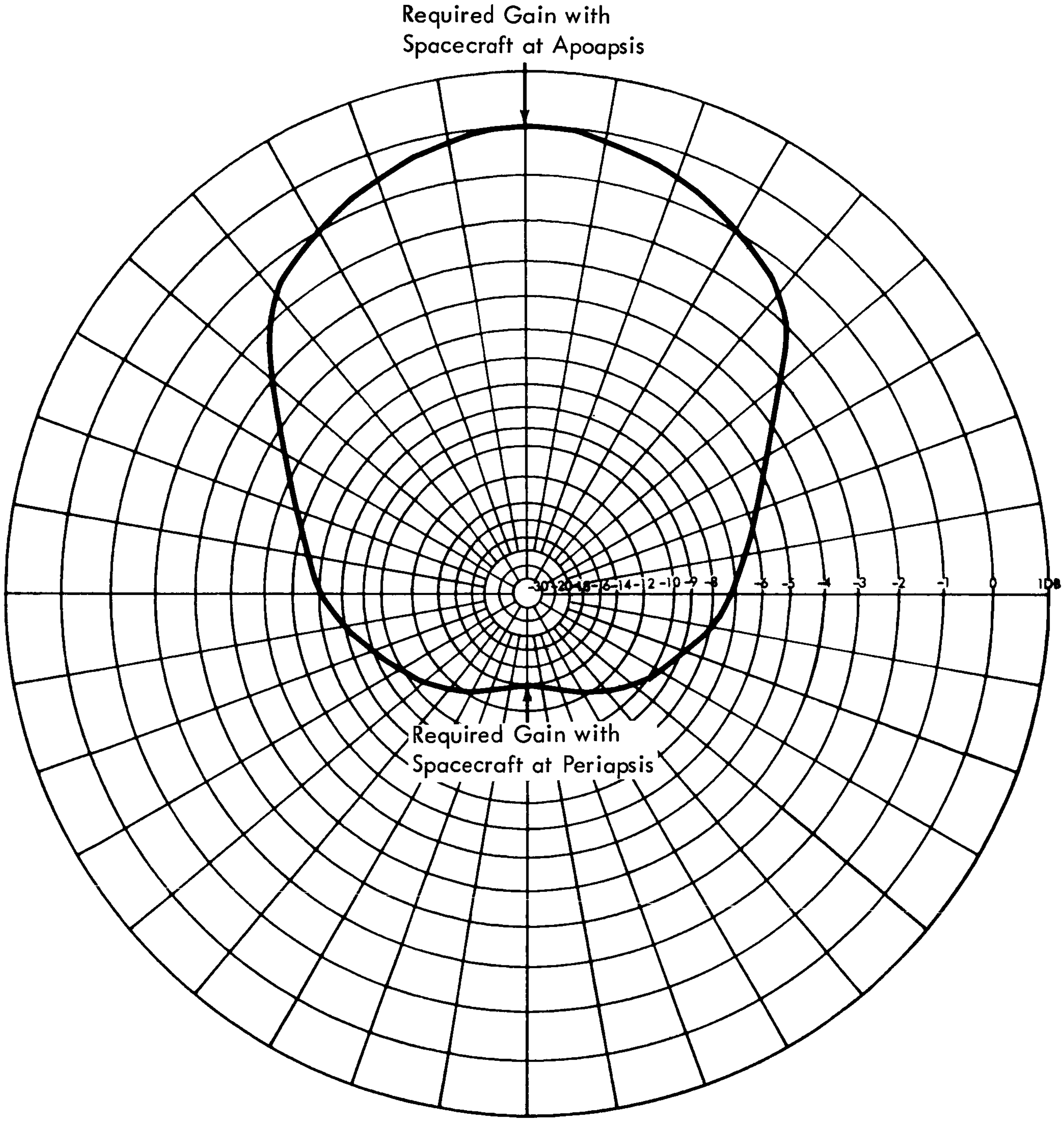


Figure 1.2.7-4: RELAY DIPOLE ANTENNA



NOTE: Spacecraft relative gain pattern required to provide 360° of uniform coverage. The plane of this pattern is the orbital plane.

Figure 1.2.7-5: DESIRED PATTERN OF DIRECTED DIPOLE RELAY ANTENNA

Table 1.2.7-1: ANTENNA SUBSYSTEM PERFORMANCE CHARACTERISTICS

LOW-GAIN ANTENNA (OMNIDIRECTIONAL)

Gain: +2.0 db peak; -6 db over 95% of sphere
Beamwidth (3-db): Omnidirectional (toroidal pattern)
Axial Ratio: 1.6 (typical, equatorial plane)
VSWR: 1.7:1
Lead Loss: - 0.9 db maximum

HIGH-GAIN ANTENNA (PRIME)

Gain: +37.0, +1.0, -0 db
Beamwidth (3-db): 2.2 degrees
Axial Ratio: 1.6 at -3 db, 1.06 on axis
VSWR: 1.5:1
Lead Loss: - 1.0 db maximum

MEDIUM-GAIN ANTENNA (BACKUP)

Gain: +24.0, +0.5, -0 db
Beamwidth (3-db): 9.3 degrees
Axial Ratio: 1.06 on axis
VSWR: 1.5:1
Lead Loss: - 0.8 db maximum

RELAY ANTENNA (HELIX ARRAY)

Gain: +13.0, +0.5, -0 db
Beamwidth (3-db): 60 degrees in orbital plane; 39 degrees out of orbital plane
Axial Ratio: 1.06 on axis
VSWR: 1.21:1
Lead Loss: -0.5, +0.0, -0.5 db

RELAY ANTENNA (DIRECTED DIPOLE)

Gain: +5.0, +0.5, -0.5 db
Beamwidth (3-db): 120 degrees in orbital plane; 100 degrees out of orbital plane
Axial Ratio: ∞ (linear polarization)
VSWR: 1.5:1
Lead Loss: -1.0, +0.5, -0.0 db

ANTENNA CONTROLLERS

Pointing Accuracy: ± 0.1 degrees in each axis
(See section 1.2.10 for additional controller characteristics)

Table 1.2.7-2: ANTENNA SUBSYSTEM WEIGHTS

ITEM	WEIGHT (Pounds)	
High-Gain Antenna		158.0
Dish and Feed	82.0	
Boom and Hinge	25.0	
Stowage Mechanisms	4.0	
Deployment Mechanisms	6.0	
Pointing Control, 2-Axis	20.0	
Secondary Support Structures	21.0	
Medium-Gain Antenna		15.0
Dish and Feed	2.5	
Boom	5.0	
Stowage Mechanism	1.5	
Deployment Mechanism	1.5	
Pointing Control, 1-Axis	4.5	
Low-Gain Antennas (2)		10.0
Antennas (2)	1.0	
Booms (2)	5.0	
Stowage Mechanisms (2)	1.5	
Deployment Mechanisms (2)	2.5	
Relay Helical Antenna		14.0
Relay Dipole Antenna		1.0
Support Mounts: MGA, LGA (2), and Relay Antennas		10.0
TOTAL		208.0

Table 1.2.7-3: ANTENNA SUBSYSTEM INTERFACES

ITEM NO.	TYPE OF INTERFACE	INTERFACE DESCRIPTION	INTERFACING SUBSYSTEM	INPUT To	INPUT From	BOUNDARY DEFINITION
1	RF	Phase-Modulated RF Signal, 2115-MHz Received and 2295-MHz Transmitted	Radio and DSIF	X	X	Cable connector at Radio Subsystem free space
2	Mechanical	HGA Support (Deployment/Gimbal Mechanism) Rotary Joint and Coaxial Transmission Line	Structural and Mechanical	X		Mounting points
3	Mechanical	Medium-Gain Antenna Mounting Provisions	Structural and Mechanical	X		Mounting points
4	Mechanical	Relay Antennas Mounting Provisions	Structural and Mechanical	X		Mounting points
5	Mechanical	LGA Booms and Transmission Lines	Structural and Mechanical		X	Mounting points
6	RF	Postseparation Capsule Data 400 MHz	Capsule and Radio Subsystem	X	X	Free space and cable connector at Radio Subsystem
7	Electrical	Command Signals for Antenna Controller	C&S and Command Decoder		X	Cable connector at C&S
8	Electrical	Position Signals to C&S and Telemetry	C&S and Telemetry	X		Cable connector at C&S and Telemetry Subsystem

Table 1.2.7-4: ANTENNA SUBSYSTEM RELIABILITY

SUBSYSTEM ELEMENTS	SINGLE THREAD	ASSESSED RELIABILITY		RELIABILITY GOAL
		REDUNDANT LGA	REDUNDANT LGA, MGA BACK-UP HGA	
Low-Gain S-band Antennas				
Low-Gain Antenna A	0.9897			
Low-Gain Antenna B	0.9897	0.99989	0.99989	
Medium-, High-Gain Antenna				
Medium-Gain Antenna	0.9924		0.99989	
High-Gain Antenna	0.9854	0.9854		
Relay Antenna (Helix Array)	0.99909	0.99909	0.99909	
Relay Antenna (Directed Dipole)	0.99910	0.99910	0.99910	
TOTAL		0.9835	0.9980	0.995

safety. Procedural safeguards will be implemented to restrict approach distances during testing. No other safety problems are anticipated.

1.2.7.6 Trade Study Summary

A detailed trade study was conducted during Task D to define the relay receiving antennas. The results are summarized in Figure 1.2.7-6.

1.2.7.7 New Technology and Development Items

No new technology or development is required for the antenna subsystem components. All necessary design represents a minor extension of current technology and design techniques.

1.2.7.8 Growth Potential

The design and coverage capability of the antenna subsystem low gain antennas is adequate for later missions. The higher data rates proposed in 1977 and 1979 flights may require a larger high gain antenna. The subsystem design and pointing control techniques are expected to be satisfactory for HGA diameters up to 20 feet.

TRADE STUDY SUMMARY SHEET	TRADE STUDY TITLE: RELAY RECEIVING ANTENNA	
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS	MATRIX OF DESIGN APPROACH	SELECTION
The relay antennas support the RF link between the capsule and spacecraft during capsule descent, entry, and postlanding periods. DESIGN CONSTRAINTS (1973 Mission) 1. Circularly-polarized antenna required at one end of relay link. 2. Entry Phase (High data rate): Requires narrow-beam antenna with maximum gain over view angle profile. 3. Entry Phase (low data rate) and Postlanding Phase: Coverage over 360 degrees in orbit plane and ±35 degrees out of orbit plane. TRADE CONSIDERATIONS Performance Simplicity Hardware Development	Design Constraints 2 and 3 require two antennas: DIRECTIONAL DESCENT ANTENNA 1. HELIX PRO: 1. Simple, proven design CON: 1. Too wide coverage out of orbit plane. 2. Must be oriented on spacecraft to suit descent. 2. HELIX ARRAY PRO: 1. Simple, proven design 2. More directivity CON: 1. Must be oriented on spacecraft to suit descent. 3. SPIRAL-FED HORN ARRAY PRO: 1. Good gain over view angle. 2. Less susceptible to spacecraft shadowing CON: 1. Awkward physical size 2. Fabrication difficulty 3. Must be oriented on spacecraft to suit descent OMNIDIRECTIONAL ANTENNA 1. DIPOLE PRO: 1. Simple, proven design CON: 1. Uneven coverage through orbit. 2. DIRECTED DIPOLE PRO: 1. Simple, proven design 2. Shaped pattern for even coverage through orbit CON: 1. Must be designed to suit spacecraft orbit	DIRECTIONAL: Performance 2, 3, 1 Simplicity 1, 2, 3 Hardware Development 1 and 2, 3 OMNIDIRECTIONAL: Performance 2 Simplicity 1, 2 Hardware Development 1 and 2 SELECTED APPROACH DIRECTIONAL: Helix Array OMNIDIRECTIONAL: Directed Dipole

Figure 1.2.7-6: RELAY RECEIVING ANTENNA TRADE STUDY

BLANK

1.2.8 Computing and Sequencing Subsystem

1.2.8.1 Design Constraints and Requirements
1.2.8.2 Functional Description and Performance Characteristics
1.2.8.3 Physical Characteristics
1.2.8.4 Interface Definition
1.2.8.5 Reliability
1.2.8.6 Trade Study Summary
1.2.8.7 New Technology and Development
1.2.8.8 Growth Potential

1.2.8 Computing and Sequencing Subsystem (C&S)

1.2.8.1 Design Constraints and Requirements

The C&S will consist of two integral units: a data processing unit (DPU) and a command path unit (CPU).

1) The data processing unit will execute all commands independent of the command path unit.

2) The data processing unit will provide time-dependent outputs to switch the flight spacecraft subsystems between their modes of operation.

3) Output time will be wired and/or set into the DPU at launch so that it is capable of executing all functions from launch to the end of the mission without the use of ground commands, except for interplanetary trajectory corrections or orbit trim maneuvers.

4) Event times in the DPU that are dependent on the particular mission, trajectory, or trajectory dispersions, will be capable of being updated by RF command.

5) The DPU will provide a time reference to the data handling subsystem and will provide a counting capability for integration of discrete signals.

6) The DPU will be capable of providing for the following flight spacecraft operations: prelaunch operations; launch operations; automatic acquisition; interplanetary cruise operations controlled on a time-dependent basis; flight spacecraft-capsule preseparation and separation sequencing, orbital insertion maneuver sequencing, and sequencing associated with solar, Canopus and Earth occultation modes; orbital operation sequencing on a mission-independent basis; planetary observation sequencing; and updating of the steerable antenna position.

7) The total state of the DPU and its outputs will be nonvolatile in the event of power transients or power dropout.

8) The logical organization of the C&S will be based on a cycled, special-purpose computer approach.

9) Timed outputs represented by the following categories will be provided by the DPU:

   a) Cumulative reference times from launch either independent of or dependent on launch date.

   b) Cumulative reference times from arrival at the planet either independent of or dependent on date of arrival.

   c) Cumulative reference times from receipt of radio-initiate command.

   d) Periodic reference times initiated at launch or by the time outputs of a), b), or c).

e) Reference time which is cyclic in nature but with varying periods initiated by on-board sensors or by the timed outputs of a), b), or c).

f) Ground command variable (prior to execution) time intervals initiated by the timed outputs of a), b), or c).

10) The DPU will have the capability to receive the following inputs from the CPU for immediate execution, or for storage and delayed execution:

a) Start time for interplanetary trajectory correction and orbit trim maneuver sequences.

b) The magnitudes and directions for planetary vehicle turns for interplanetary trajectory correction and orbit trim maneuvers, for flight spacecraft-flight capsule separation attitude or orientation, and for the orbital insertion maneuver.

c) Magnitude of the velocity increments to be provided by the propulsion subsystem for the interplanetary trajectory correction and orbit trim maneuvers.

d) Start times for capsule preseparation maneuver, for the capsule separation sequence, and for the orbital insertion maneuver sequence.

e) Timed event sequences associated with orbital operations.

11) The CPU will decode and execute functions independent of the DPU. The CPU will minimize command acquisition time with the radio subsystem.

12) The CPU will be capable of updating, through the C&S, initiate times of all events.

13) The CPU will provide all maneuver command words to the DPU for storage.

14) The CPU will command backup for events initiated by the DPU.

15) The CPU will provide command words to the DAE to turn on or off various science instruments and recorders.[2]

16) Data rate between the CPU and DPU will be 6.4 kc.

17) Data rate to DAE from the C&S will be 4.8 kc.

18) Command bit rate from the radio subsystem to the CPU will be 1 bit per second.

19) Unreferenced end-circuits will be employed in the C&S as necessary to avoid circuit return paths through circuit wiring and/or electrical ground net. A number of unreferenced end-circuits sharing a common circuit return need not have individually isolated return terminals.

20) The return terminals of referenced end-circuits will be connected to the circuit common for the containing equipment.

[2] In Vol. 4, Section 4.0, it is recommended to incorporate the data automation equipment (DAE) command functions in the C&S. The C&S would then interface directly with the science subsystem, providing time, sequencing, on-off control and scan platform control.

1.2.8.2 Functional Description and Performance Characteristics

The computer and sequencer subsystem as shown in Figure 1.2.8-1 consists of two data processors, an interprocessor control unit, two oscillator/registers, two command path units (CPU), and a redundant signal conditioning unit.

Data Processor Unit -- Two data processors (Figure 1.2.8-2) will execute the flight program simultaneously and synchronously under normal operating conditions. Error detection and correction techniques are incorporated to provide for failure mode operation. Parity checking and generation for all words "read from" or "written in" memory, as well as interprocessor word comparison, is provided to give essentially a programmed majority vote capability for information transfers. The data processor program also features diagnostic and self-checking routines to inhibit erroneous outputs. A basic internal bit rate of 6.4 kc was selected to accommodate the data processing rate of the mission sequences. The memory is a random access, by word, and serial-by-bit magnetic core system.

All commands from mission control to the DPU will go through the command path units (CPU). On receipt of a valid command for the DPU, the CPU will generate a data ready signal. The data ready signal will cause the DPU to interrupt and switch to a special subroutine to accept incoming commands. Once the command acceptance subroutine is entered, the DPU will shift the command into both processors. The incoming commands are stored in memory locations specified by the subroutine. The commands may be either of the following types:

1) Immediately Executed Commands -- The DPU will initiate execution of these commands within 1 second. These may be discretes, timed angular maneuvers, magnitude integrations for velocity maneuvers, or antenna or platform positioning commands.

2) Stored Program Commands -- These commands perform the flight program sequences, updating of spacecraft time, telemetry outputs, and error detection and correction routines.

The DPU must perform a large number of discrete commands (as identified in Section 1.2.8.4) to fire squibs, switch guidance and control functions, and to sequence telemetry modes. Timed angular commands are executed to perform maneuvers about the roll, pitch, or yaw axis. Magnitude commands are used to perform velocity maneuvers causing the firing of the midcourse engines and integration of the accelerometer for velocity change.

The DPU provides updating commands for the high gain antenna throughout the mission. These commands contain angular position and a periodic time for updating the antenna. During the mission, antenna pointing commands will be executed first by comparing the antenna position obtained from a digital shaft encoder, and then issuing the appropriate direction signals based on this comparison. Then the periodic time value will be added to spacecraft time for the next updating of the antenna. When the time interval has elapsed, the angular value of the command will be incremented 0.2 degree, and the new angle commanded.

At launch, the DPU will be loaded from launch control at 200 words per second through the CPU. The DPU will receive a load signal from launch control. A fixed-wired program will load and then readout the entire memory content for verification.

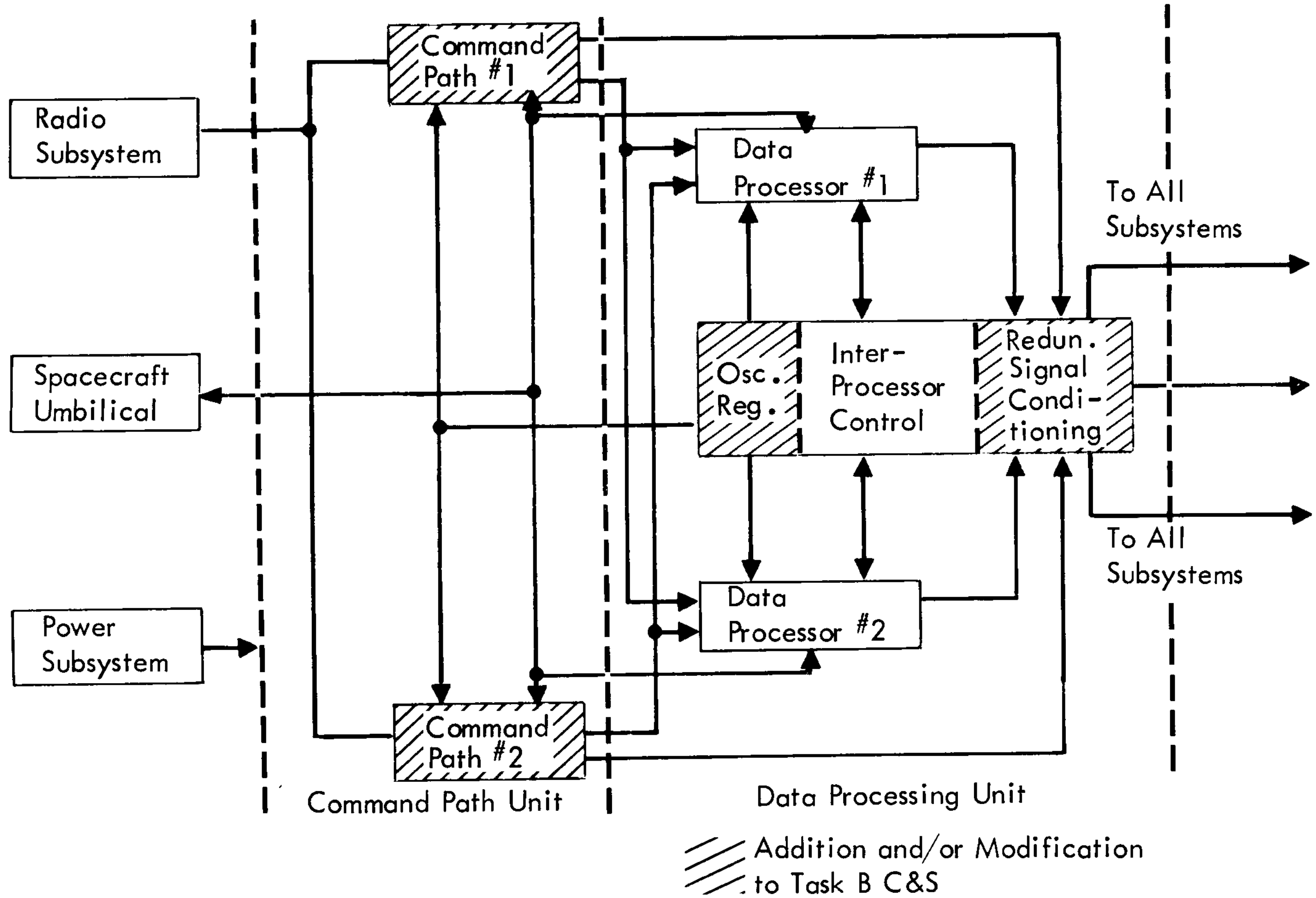


Figure 1.2.8-1: COMPUTER AND SEQUENCER SUBSYSTEM

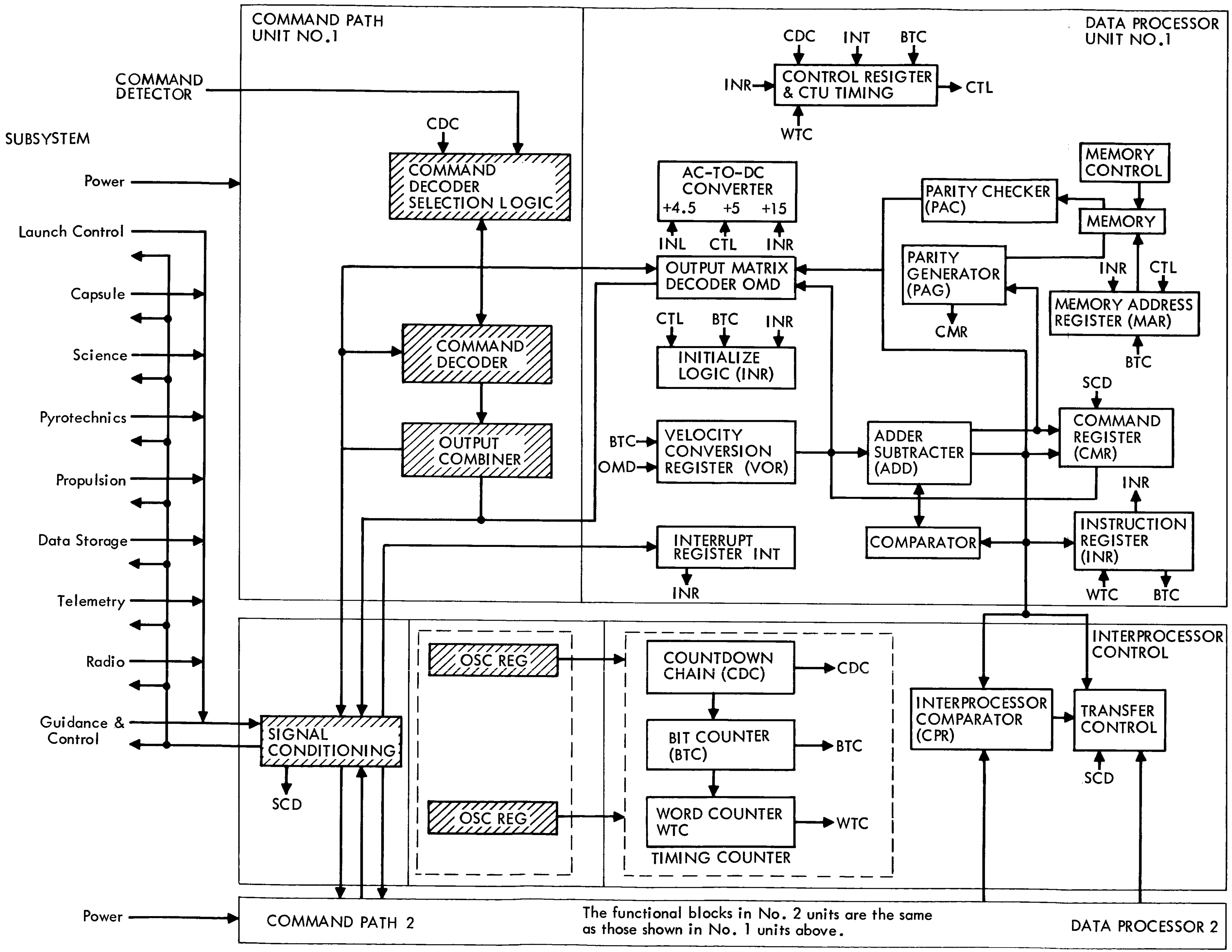


Figure 1.2.8-2: COMPUTER AND SEQUENCER FUNCTIONAL BLOCK DIAGRAM

The DPU incorporates provision for special interrupt functions while executing the main flight program. There are 11 interrupts presently required to be executed; some are associated with the flight program, such as spacecraft time and telemetry. Four interrupts, Sun occultation, Canopus occultation, limb crossing, and terminator crossing, are implemented to provide initiation times for orbital events. The spacecraft time is recorded automatically under program control within the DPU whenever these events occur. Then, by program subroutine, orbital event sequences may be initiated from this recorded time.

Spacecraft time is derived from the oscillator/register and accumulated in each processor in three locations in the memory. Time is accumulated in 1-second increments to provide a total time capacity of 776 days. Spacecraft time is accurate to 1 part in 10^6 over the transit interval between Earth and Mars, and provides accurate arrival time for orbit insertion of $\pm$ 20 seconds without ground updating. Each processor is programmed to provide error detection and correction of spacecraft time, using internal parity and past event checking routines as well as interprocessor data transfer for correction. The DPU provides timed outputs, on a time-dependent basis, to flight spacecraft subsystems for switching systems between various modes of operation.

Command Path Unit -- Figure 1.2.8-3 shows the functional block diagram for the preferred command path unit design. Each command path unit decodes the received commands from the command detector in the radio subsystem and provides independent command word outputs to the DPU and DAE subsystems. Redundant implementations with dual outputs are provided to avoid a single-thread path for critical DPU update functions, such as trajectory corrections.

Each CPU also provides a real-time backup decoding and executing capability independently of the DPU. The redundant decoded outputs are combined in a command output combiner to reduce interface complexity and cable weight. A single-thread interface with the other spacecraft subsystems is permitted because the command decoding function of the command path units are primarily backup for commands that will normally be executed by the DPU subsystem. A dual-thread interface for critical functions can be implemented for a penalty in command output combiner weight and external wiring weight.

A separate power conditioner is provided for each CPU to prevent a single power supply failure from causing subsystem failure. Each power conditioner has the capability to operate from either power bus so that a power bus failure does not remove subsystem redundancy.

Signals from a command detector located in the radio subsystem enter the detector selection logic. When an inlock signal is presented to the CPU, the control logic permits data to enter the command register (unless internal command processing or execution is occurring). The use of this inlock command data gate thereby inhibits command processing if command modulation is lost. As input data is shifted through the command register, the 30-bit AND gate continuously looks for the presence of word sync in the input data. Upon detection of word sync by this gate, the control logic resets the command register to all zeros except for the input stage. During the remainder of the command verify and execute period, word sync signals are ignored by the control logic. The control logic detects this overflow and inhibits further input data until verification and execution (if verified) are accomplished. In this manner, the command register is time-shared and used as a ring counter during the data fill mode. This eliminates the need for a Modulo counter in the control logic. Parity is checked on the actual contents of the command register

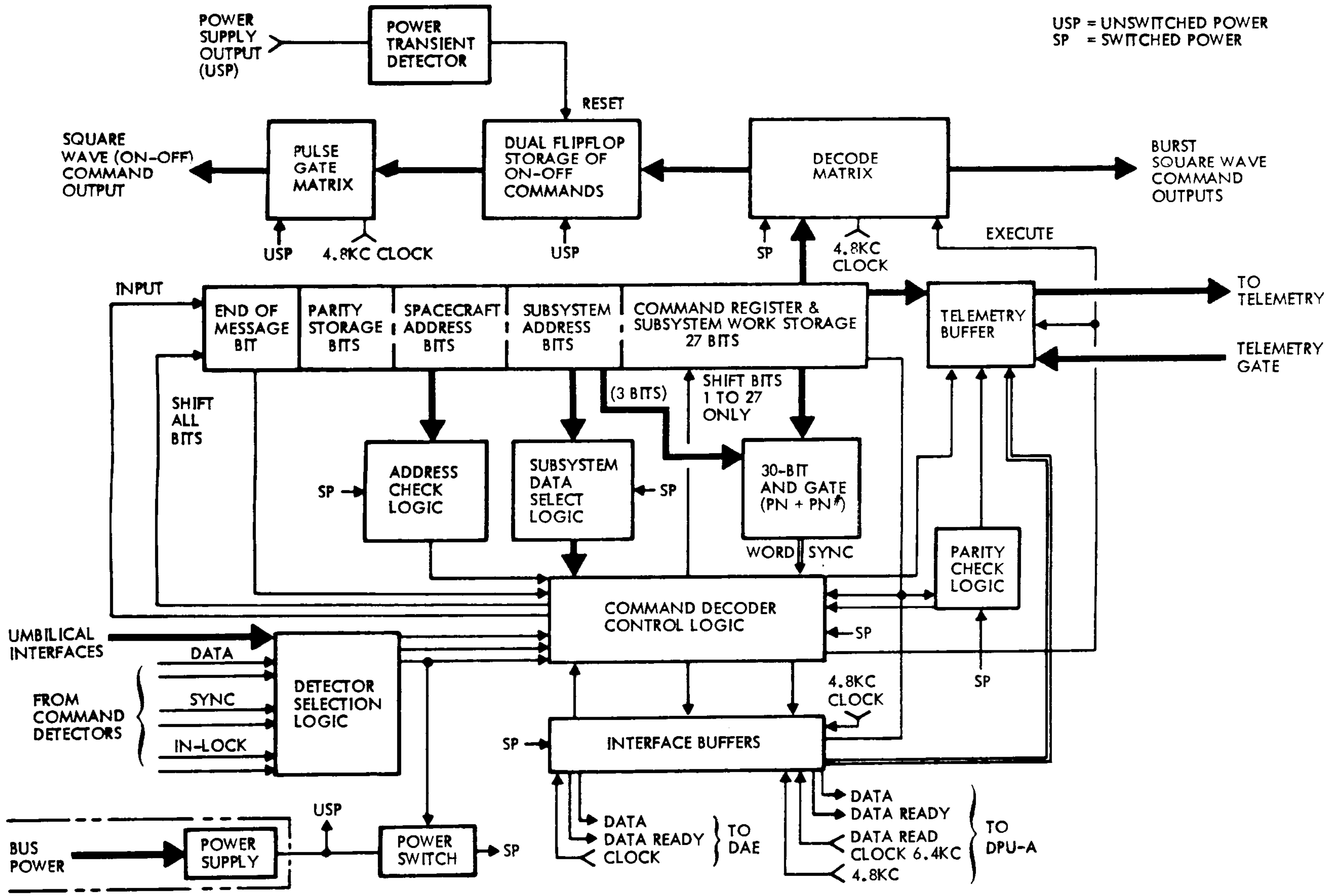


Figure 1.2.8-3: COMMAND PATH UNIT

rather than on its input to prevent a failed register from receiving correct data which would then be sorted and executed falsely. Register-contents-parity-checking is performed by circulating the command register data twice through the control logic while at the same time the parity check logic operates on the serial data. The contents are also loaded into the telemetry buffer during the second circulation. An execute signal is generated by the control logic for real-time command initiation if (1) the data passes the parity check, (2) the address check logic verifies that the command was intended for the subject spacecraft, and (3) the real-time command is verified. Alternatively, if parity and spacecraft address check, "data-ready signals" are generated if the data is intended for the DPU or DAE subsystems. The control logic performs these functions, using inputs from the subsystem data select logic.

For data to be decoded by subsystems other than the command path unit, the data-ready signal causes the DAE or DPU to transmit clock pulses to the interface buffer. When appropriate, the control logic uses these clock pulses to shift bits 1-27 from the command register through the interface buffer to the appropriate subsystem. The data transfer operation occurs in less than 1 second (a bit time), so that no forced time-gap constraint exists between successive commands.

Oscillator/Register -- The computer and sequencer's (C&S) CPU and DPU derive all operational times from the oscillator/register unit. This block consists of two precision crystal oscillators and the associated countdown registers cross-coupled together feeding pulses to the triplicated majority countdown chain. Failure in one oscillator/register will not affect the operations of the other.

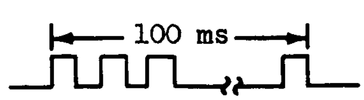
Signal Conditioning -- All inputs and outputs for the C&S are processed through the signal conditioning unit. This unit provides interface compatibility between the C&S and other subsystems (refer to section for interface definitions).

The performance characteristics of the C&S are summarized in Tables 1.2.8-1 and 1.2.8-2.

The C&S electrical interfaces with other spacecraft subsystems are shown in Figures 1.2.8-4 and 1.2.8-5.

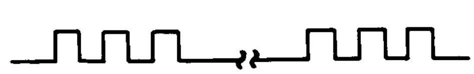
The C&S will provide (and accept) four different types of electrical signals to the spacecraft subsystems. These electrical inputs and outputs are transformer-coupled to provide unreferenced end circuits and to eliminate multiple ground return paths. The a.c. signals are integrated to obtain a discrete level and to provide noise immunity. The four types of signals are described in the following paragraphs.

1) Burst Square Wave



A 6-volt, 4.8-kc pulse train with a duration of approximately 100 milliseconds. Signal is used for interfaces that involve electro-explosive device firing, and for those requiring signals to drive latching devices such as flip-flops and latching solenoids.

2) Square Wave (On-Off)



A 6-volt amplitude uniform square wave with a frequency of 4.8-kc. The square wave is either present or not present to provide a signal similar to a discrete level voltage change.

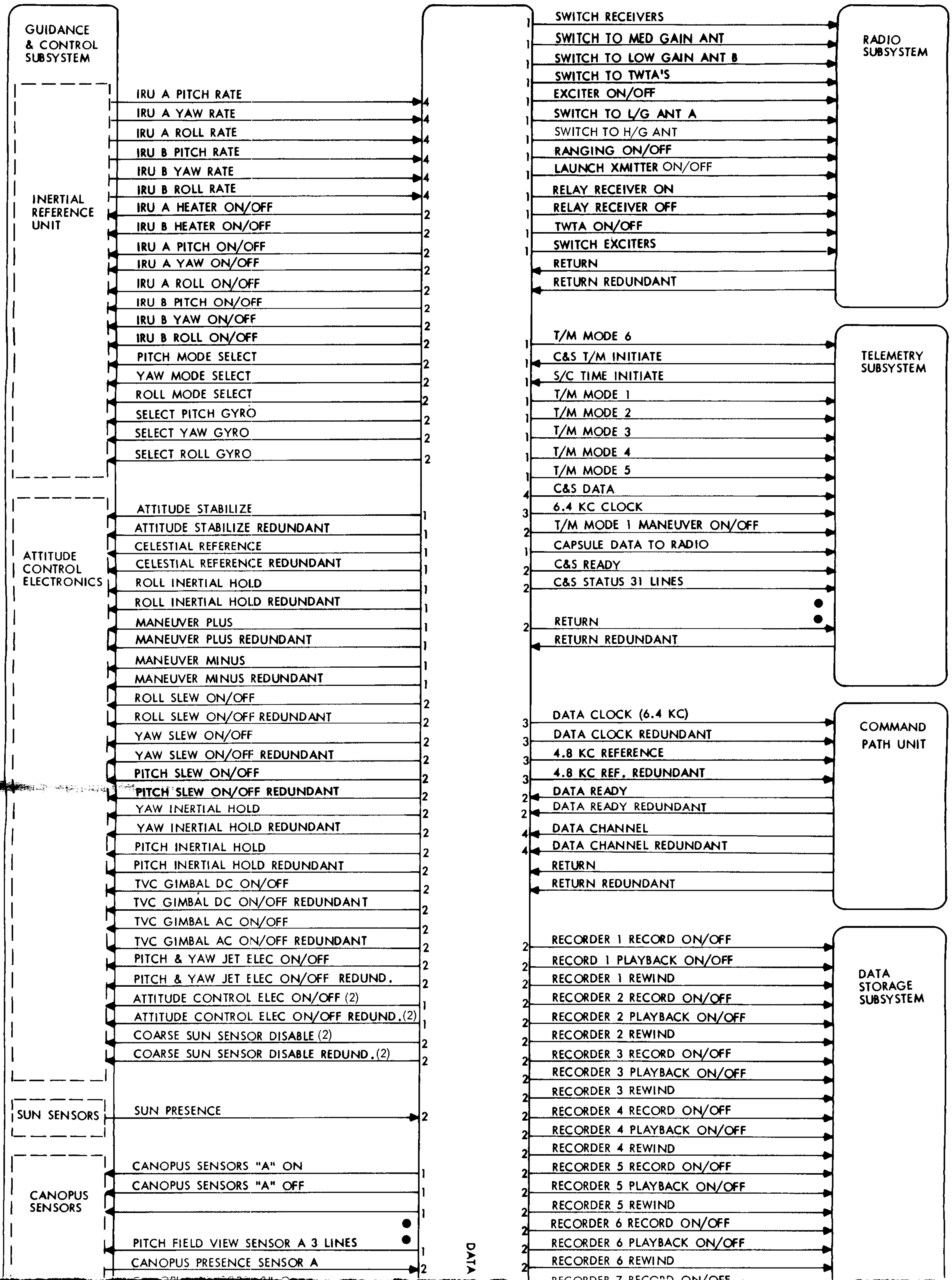
Table 1.2.8-1: DATA PROCESSING UNIT PERFORMANCE CHARACTERISTICS

PARAMETERS	PERFORMANCE CHARACTERISTICS
Reliability	0.9614
Memory — each (2)	
• Capacity	1024 words (27,648)
• Type	Multiaperture Core
Command Capability	140
Bit Rate (Operating)	6.4 kc
Word Length (External and Internal)	27 bits
Word Rate	1 word/5 milliseconds
Program Modes	Immediate Execution Stored Program
Program Routines	
• Immediate Execution Modes (Highest Priority)	Magnitude computation, comparison and execution
	Discrete execution jump (instruction sequence modification)
	Stored program address execution
	Telemeter memory
• Stored Program Mode	Magnitude computation, comparison and execution, discrete execution, wait time computation, comparison and execution
	Jump (instruction sequence modification)
	Compare time comparison and execution
• Interrupt Modes	Store Sun occultation time
	Store Earth occultation time
	Store Canopus occultation time
	Command path unit interrupt
	Spacecraft time computation
	Telemeter C&S data
	Terminator crossing time
	Limb crossing time
	Inertial reference
	Separation
	Load memory & reset clock
Storage Modes	
• Automatic Sequential Storage	1 command words/stored word
• Random Access Storage	2 command words/stored word
Computation	
• Word Addition	1 word/5 msec
• Serial Comparison	26 bits/5 msec
• Magnitude Addition	10 bits/second
Spacecraft Clock	
• Type	Binary
• Capacity	776 days, 17 hours, 37 minutes, 44 seconds
• Least Significant Bit	1.0 second
• Accuracy	1 part in 10^6 in 215 days
• Reset Modes	To Zero - manual, total count, and command
	Time - command
Velocity Conversion Register Input Rate	600 pps
Telemetry Memory Storage	16 (26-bit words)/processor

Table 1.2.8-2: COMMAND PATH UNIT PERFORMANCE CHARACTERISTICS

PARAMETERS	PERFORMANCE CHARACTERISTICS
Approximate number of commands	123 burst squarewave outputs 60 squarewave (ON-OFF)
Command word length	Same as Task B
Sync word length	30 bits
Number of addressable spacecraft	4
Error detection	Parity, plus special coding for real-time commands
Subsystem reliability assessment	0.959
Command path unit reliability assessment.	0.9977
Internal clocking rate	4.8 kbps
Probability of false command per command (executed by command path unit) with bit error rate of 10^{-4}	10^{-20}
Probability of command word error to DPU or DAE with bit error rate of 10^{-4}	10^{-16}
Growth capability	12 factorial commands
Command bit rate	1 bit per second
DAE and DPU word length	27 bits
Output interfaces	4.8 kc clock trains (none discrete level) 6.4 kc clocking rate to DPU
Failure modes	No single failure can cause a false command

BLANK



FOLDOUT FRAME 1

1-227

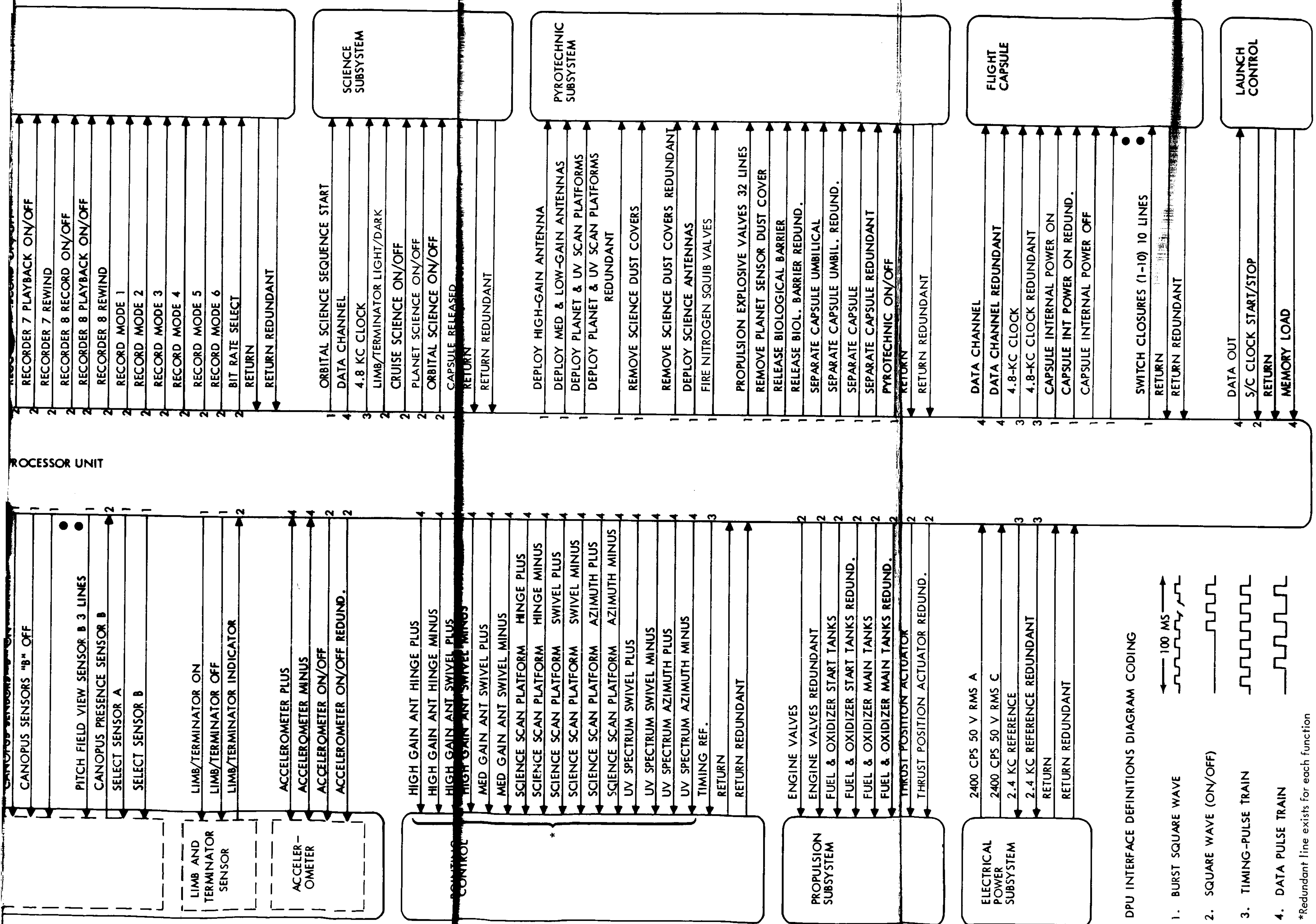
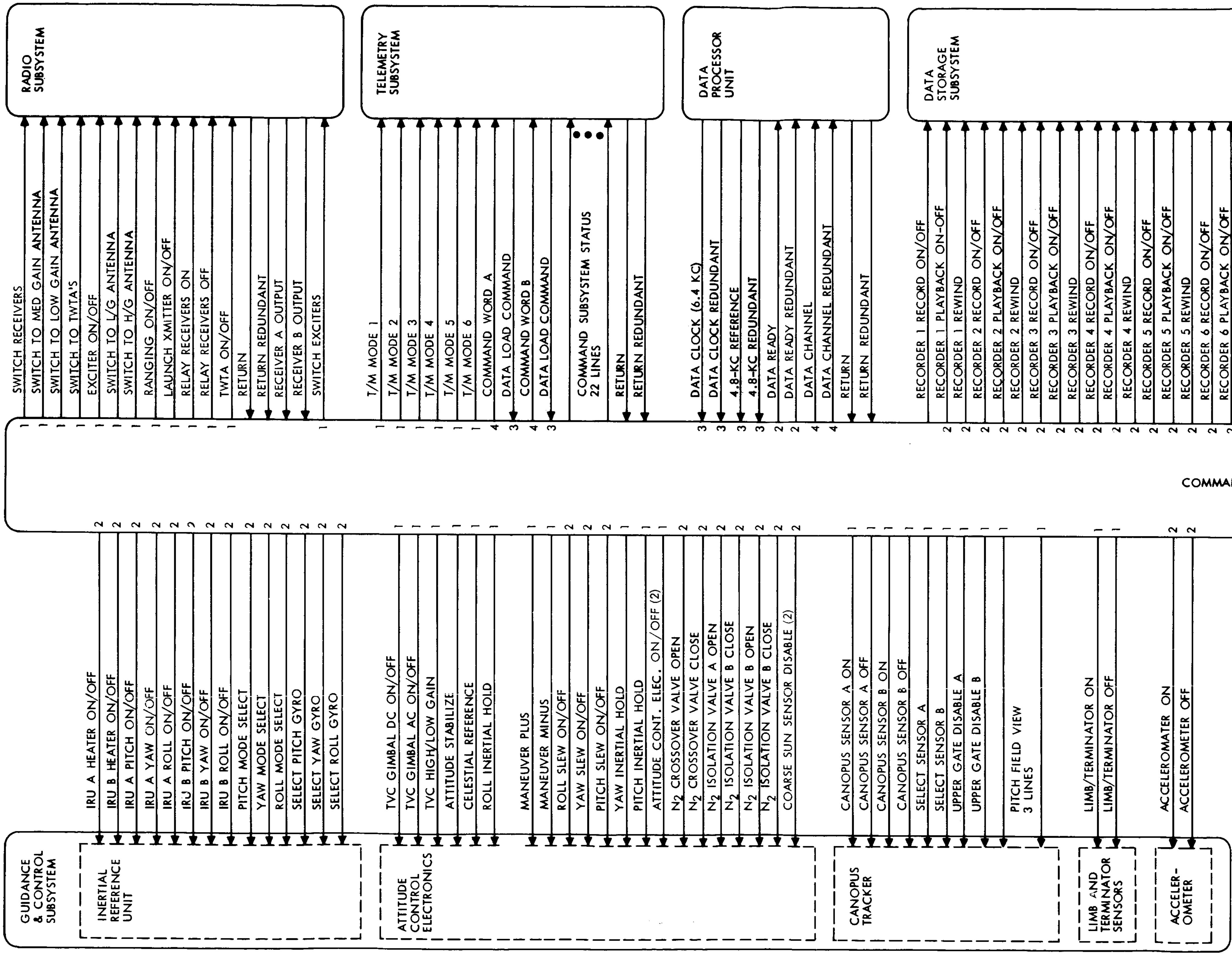


Figure 1.2.8-4: DATA PROCESSOR UNIT ELECTRICAL INTERFACE

FOLDOUT FRAME 2

FOLDOUT FRAME 3



FOLDOUT FRAME 1

1-229

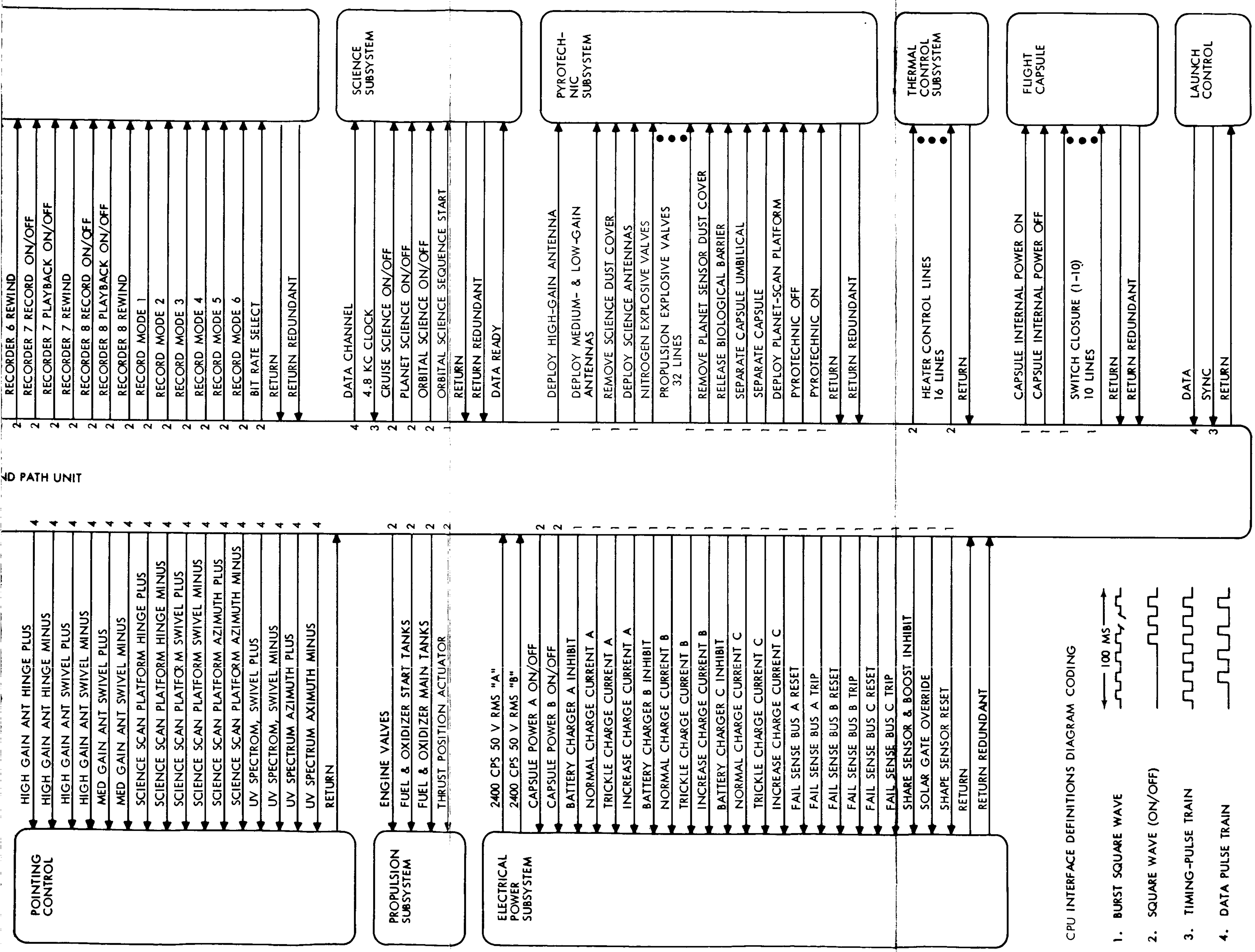


Figure 1.2.8-5: COMMAND PATH UNIT ELECTRICAL INTERFACE

3) Timing Pulse Train 

A 6-volt amplitude clock reference pulse train of uniform pulses used for providing clock timing trains of desired frequencies and for synchronization.

4) Data Pulse Train 

A 6-volt amplitude pulse train of binary information for data transmission between the C&S and other subsystems.

Figure 1.2.8-6 shows the preferred design electrical output function interface configuration of the C&S with another typical subsystem. The CPU has fail-safe redundant coding and output logic, combined in a single-output transformer for each output function. The DPU has fail-safe redundant coding and output logic in each processor for critical output functions, e.g., propulsion ignition and orbit insertion maneuvers. This configuration ensures that a single failure will not cause a false command to be issued, nor prevent the sending of a true command.

For noncritical output commands (e.g., some appendages deployment, and certain mode switching), a single output circuit in the DPU is controlled from either processor. The two on-off type square-wave outputs from the C&S subsystem are synchronous and of opposite phase. In the event of an "on" failure in one subsystem, the other can still control the output function by appropriate use of the biphase signals through "exclusive or" logic in the receiving subsystem.

1.2.8.3 Physical Characteristics

The C&S physical characteristics are summarized below:

Weight	68.4 pounds
Power	68.8 watts average 78.4 watts peak
Volume	3708 cubic inches

1.2.8.4 Interface Definition

The C&S has electrical, thermal, and structural interfaces with other spacecraft subsystems. Mechanical and thermal interfaces are controlled by packaging design, and are summarized below:

Thermal -- Radiation and conduction heat transfer to maintain operating temperature of the C&S within limits of $35^{\circ}F$ to $95^{\circ}F$.

Structural -- Mechanical interface to provide structural support and mounting.

Electrical interfaces are shown in Figure 1.2.8-4 and 1.2.8-5.

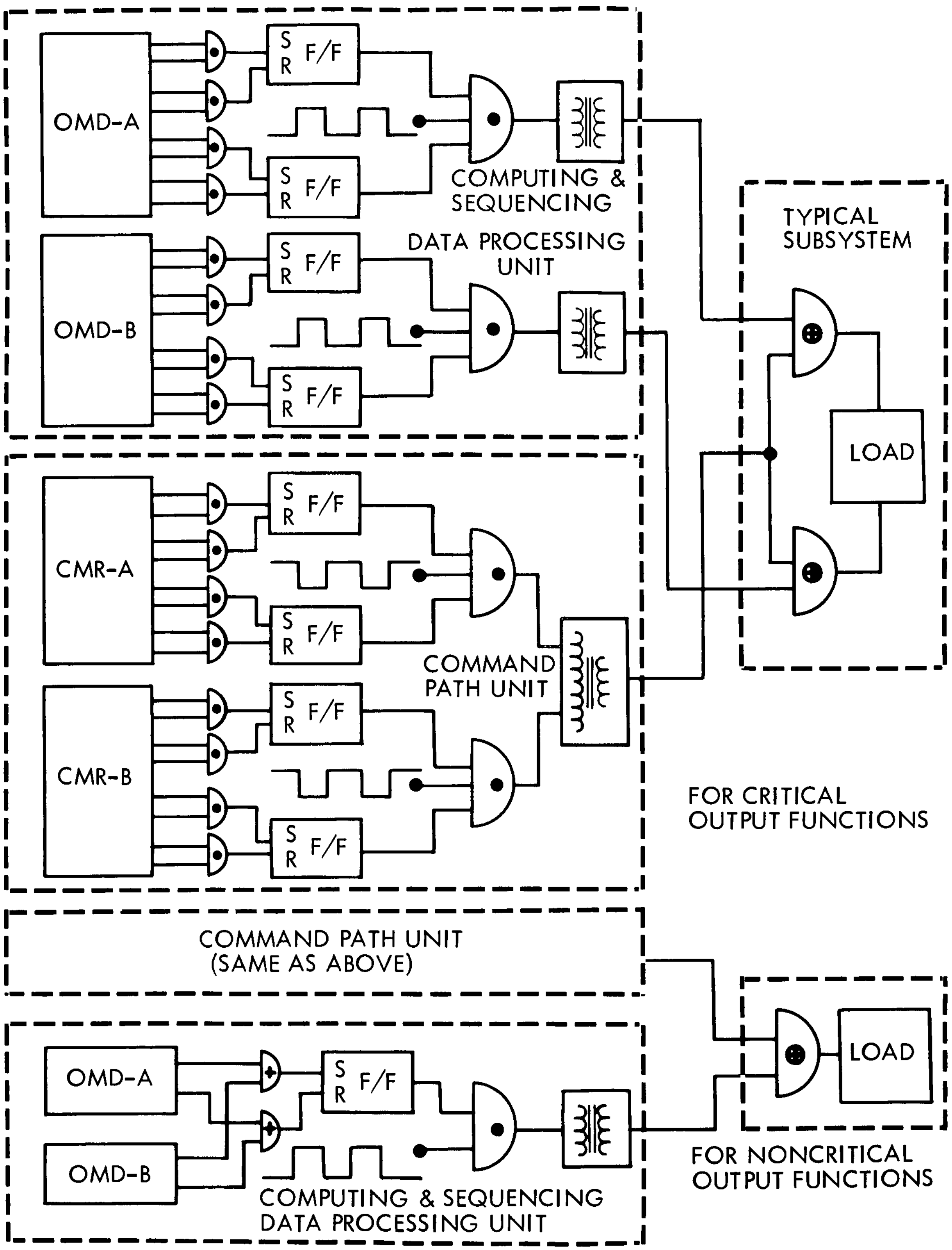


Figure 1.2.8-6: COMPUTING AND SEQUENCING OUTPUT FUNCTION INTERFACE

1.2.8.5 Reliability

The assessed reliability is shown in Table 1.2.8-3.

Table 1.2.8-3: RELIABILITY ASSESSMENT
(Computing and Sequencing Subsystem)

SUBSYSTEM ELEMENTS	RELIABILITY ASSESSMENT		RELIABILITY GOAL
Data Processor		0.9614	
Signal Conditioning & Processing	0.9739		
Output Matrix Decoder	0.9976		
Comparator	0.9986		
Memory	0.9909		
Interprocessor Comparator		0.9999	
Triple Majority Vote	0.9999		
Interprocessor Comparator	0.999991		
Transfer Control	0.999998		
Oscillator/Register	0.999998		
Command Path Unit		0.9977	
Selection Logic	0.99995		
Command Decoder	0.9978		
Output Combiner	0.99995		
Total Subsystem		0.959	0.942
Mission critical functions that can be controlled by either the DPU or the CPU		0.9999	

The C&S mission reliability is achieved by using redundant processors, with self-checking and error-detecting programs, with the interprocessor control to permit the transfer and comparison of information between processors and redundant real-time command paths. When the two DPU's are operating out of synchronization as a result of transients, an operational case of particular interest occurs. Internal self-checking and error-detection programs cannot distinguish between the data processor that is operating on the correct segment of the mission profile and the one which is in error. Resynchronization will be accomplished by ground commands issued through the CPU. With the exception of this single error, the preferred C&S design is capable of meeting the design requirements of:

- CPU & DPU functional independence
- The ability to perform the mission with single catastrophic failures in either the CPU or DPU or both.

<u>Data Processing Unit</u> -- The DPU employs a cooperative parallel-path type of redundancy with failure detection and passivation capabilities to achieve the allocated reliability through hardware and software techniques.

Specifically the DPU is designed to:

1) Passivate single failures in output logic and interface circuits through redundancy.

2) Detect internal processor faults through parity and comparison tests.

3) Initiate self-diagnostic tests in each processor to determine, and in the majority of cases to correct or work around, the faults through software.

4) Exercise those signal paths critical to the performance of mission-critical functions prior to execution to ensure readiness.

5) The nonsync error mentioned above which cannot be corrected or diagnosed on-board will place the DPU into the halt mode and be corrected by commands through the CPU.

Command Path Unit -- The CPU employs a parallel-path type of redundancy. The unit functions primarily as a backup for the DPU in command execution and provides a means of accumulating and verifying ground-transmitted data prior to passing the command to the DPU for processing. Failure detection and passivation in the CPU is strictly passive; the philosophy is not to execute the stored commands and to initiate real-time commands as a workaround. The parallel path redundancy ensures functional performance with a single failure. The preferred design within each path also utilizes forced shift of data and redundant coding to prevent false command actuation by spurious signals and transients.

Oscillator/Register -- The oscillator/register is made dual-redundant. Normally, both paths are operating, feeding the CPU and DPU. However, the two paths are cross-coupled so that the failure of one, which will result in a d.c. output will not affect the other. The clock pulses are a.c. coupled through transformers.

The C&S as configured will be capable of performing the mission with a minimum of one failure in each unit without degradation. The DPU in selective modes can still operate with multiple failures.

Failure Detection -- Each processor parity-checks every word read out of memory. After parity check, each word is compared with the corresponding word from the other processor in the interprocessor comparator (CPR). For critical functions, errors in loading the command register (CMR) will be detected by including a compare instruction in the subroutine which circulates the word through the CMR while comparing it with the contents of the memory location from which it originated.

Identifiability and Performance Effects of Single Failures in the Data Processors

1) Failures in Function Decoding and Output Interface -- The signal path from command register through output matrix decoder (OMD) to the output function interface is sufficiently redundant to operate correctly in the presence of a single fault (permanent or transient). The command word in the CMR presents the coded command in two identical 10-bit locations. Each 10-bit code is decoded in the OMD by independent logic chains and the result combined in an AND gate. This function insures against single failure misfires since a command cannot be issued without agreement between the redundant logic paths. Failure to issue the command is avoided by providing the same output function to the subsystem load from the second data processor.

Assuming a single failure, one of the processors will supply the correct output and the failed processor will have no output.

2) Memory Location or Parity Generator Failure -- A single failure in a memory location or in the parity check on readout and identification of the failed processor is immediate. It will be passivated by automatically switching to a HALT mode.

3) Permanent Hardware Failures in Other Locations -- Permanent hardware failures in locations other than the output gating will produce discrepancy between processors in program sequence or data word content. The discrepancy will eventually be detected by the CPR which will initiate self-diagnostic tests by each processor. In most cases the presence of a permanently failed component can be detected, the failed processor can be halted, and the healthy processor returned to its normal program. The diagnostic routine is one that examines the present value and two past values of time for the correct magnitudes. The routine uses the instructions ADD, SUBTRACT, COMPARE, BRANCH, and JUMP. In addition, the interrupt instructions required to detour the program through the test and return to the point of interruption are exercised. This diagnostic test will detect failures in all registers (except VCR and MAR) since they must all shift while loading serially. Much, but not all, associated gating is also tested.

   Some component failures cannot be detected by this diagnostic test. Examples are:

   - Control register (CTL) gating associated with instructions not used in the test.
   - Failure of memory access to locations not used in this test.

4) Transient Failures -- A much larger class of single failures that will not be identifiable are those due to transient effects. If one bit of an instruction or address word for example were lost due to some transient effect, the two processors could easily arrive at different program addresses or simply differ on the content of a data word. The discrepancy would be signalled by the CPR, each processor would execute its own diagnostic test, and no malfunction would be detected. In such cases of single transient failures, no evidence remains in the processors that can distinguish between the disturbed and healthy processors. The difficulty can be resolved in all cases by a third synchronous computer located at the SFOF.

Recommended Configuration and Operating Procedure -- Possible design approaches are listed on the trade summary sheet. Only design approaches No. 3 and No. 5 have sufficient redundancy to completely satisfy the minimum success requirements. However, these approaches carry a heavy penalty in power, weight, and volume compared to the recommended approach (No. 2). As described above, the two synchronous processors (approach No. 2) can automatically compensate for the effects of a large proportion of the possible single-component failures. Furthermore, with the addition of certain command procedures, the probability of successful operation can be significantly improved. Therefore, the failure criterion was relaxed with respect to the use of commands. Specifically, reliance on the command path will be accepted for the special failure cases that cannot be handled automatically by two cooperative synchronous processors.

For critical functions, particularly orbit insertion, special operating procedures are required. Backup commands for initiation and termination of critical functions will be transmitted and timed to take effect via the command path within the C&S just after the processors should have executed the corresponding commands. When processor discrepancy is detected and one failed processor cannot be identified, both processors will switch to passive state (no outputs to other subsystems) leaving the command path free to execute the scheduled functions. Other operating procedures include special tests that will be inserted into the processor program to verify the processor's ability to perform an anticipated critical function. For example, before execution of orbit insertion maneuvers, the velocity conversion register (VCR) will be operated and a sum check performed on all memory locations used by the subroutines that accomplish the maneuvers.

These tests will exercise all registers used for the critical functions. The tests are designed to produce arithmetic results which are compared with stored constants. Processors failing the test are automatically switched to passive state. During noncritical mission periods, a ground-based third processor (or equivalent simulation) will be used to evaluate the performance of the two on-board processors as reported by telemetry. The command path will be used to carry out error detection and error correction procedures as required by the on-board processors.

The strategy for achieving reliable operation with design approach No. 2 is summarized as follows:

1) Output logic and interface circuits are sufficiently redundant to meet the mission success criteria.

2) In the remainder of the processor, most failures will be detected by interprocessor comparison. Memory failures will be detected and identified by parity checking.

3) Failure detection initiates self-diagnostic tests in each processor, and in the large proportion of possible single failures the failed processor will be identified and automatically switched to passive state.

4) Special error detection tests will examine processor readiness to perform critical functions and a failed processor will be automatically passivated.

5) Some single failures can produce interprocessor discrepancies that cannot be analyzed by the on-board processors. Both processors will then automatically switch to passive state and the command path must be relied on to execute critical functions.

6) During noncritical mission periods, use of the command path will permit analysts to carry out error correction as well as error detection procedures. Because of the great flexibility of reprogrammable on-board computers, many failures can be compensated for that might otherwise be catastrophic.

1.2.8.6 Trade Study Summary

The following trade studies were performed:

Data Processor Configuration -- The summary of configuration trades for the data processors is shown in Figure 1.2.8-7.

Destructive Readout versus Nondestructive Readout Memory -- Nondestructive readout memories are only now becoming available and none are as yet space-qualified. Destructive readout memories, on the other hand, present no technical problems and are presently space-qualified. The trade summary is shown in Figure 1.2.8-8.

Sequential versus Random Access Memories -- A sequential access memory would require higher speed operation than is presently available or space-qualified. Random access memories present no technical problems and space-qualified units are available. The trade summary is shown in Figure 1.2.8-9.

1.2.8.7 New Technology and Development

No new technology or development is required for the computing and sequencing subsystem.

1.2.8.8 Growth Potential

The C&S subsystem with its computer-like organization is not mission-dependent. Variances in mission objectives and mission profile sequence can be accommodated without hardware modifications. The preferred subsystem units described above have the following growth factor:

DPU:	Storage:	1024 words; 700 used; 300 available	
	Command:	140;	Growth 30%*
CPU:	Command:	140;	Growth to 200

Maneuver accuracy and spacecraft clock accuracy can also be increased without appreciable hardware changes. The only impact would be the use of a higher frequency out of the oscillator/register and an increase of approximately 4 watts to the DPU memories.

---

* The growth figure indicated is the available capability in the present design without any modification or additions. The inclusion of the DAE functions as recommended in Vol. IV requires approximately 150 additional discrete commands. It will be necessary to include at least seven additional designator decoders. If necessary, further addition of designator decoders will give a growth capability of 600 commands without a change of basic concepts.

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE: DATA PROCESSOR CONFIGURATION TRADES MATRIX OF DESIGN APPROACH					SELECTION
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS		1 2 Independent Parallel Synchronous DPU	2 Dependent Parallel Synchronous DPU	3 Triplicated Majority DPU	4 DPU with 1 Operate - one standby	5 Single Quadded Logic DPU	
No single failure of electronic or electrical parts or component shall cause a catastrophic effect on the mission. The DPU shall be capable of performing all critical mission functions with a single failure without mission degradation and without the use of commands for mission control if the spacecraft follows a nominal trajectory. The total state of the DPU and its outputs must be nonvolatile in the event of power transient or power dropout. The parameter considered in the order of importance are: 1) Meeting Criteria 2) Reliability 3) Power 4) Weight and Volume 5) Cost or parts Count 6) Design Complexity *Design approach No. 2 does not meet the "without use of commands" part of the above criterion. A small portion of possible failures can be compensated for only if use of commands is accepted as an automatic back-up procedure.		1) This configuration cannot meet the requirement for no single-failure catastrophe because one critical output command may fail "ON" and cause loss of the mission. 2) Reliability: 0.99 3) Power: 52 watts 4) Weight & Volume: 45 pounds 2560 cu. in. 5) 1300 Integrated circuits 2800 Discrete Parts 2 Memories 6) Design complexity factor of one -- equivalent to a single non-redundant system	1) Provides for internal selfcheck and interprocessor crosscheck but meets criterion only partially*. 2) Reliability: 0.959 3) Power: 52 watts 4) Weight: 45 lb 2560 cu. in. 5) 1400 Integrated Circuits 2800 Discrete Parts 2 Memories 6) Design complexity factor of one with more programming.	1) Meets all requirements. 2) Reliability: 0.972 3) Power: 85 watts 4) Weight: 85 lb Volume 3840 cu. in. 5) 2800 Integrated Circuits 2800 Discrete Parts 3 Memories 6) Equivalent to Configuration No. 1.	1) This configuration cannot be tolerated. If the operating unit fails, the standby unit has no method by which to determine the program sequences and the required reference time. 2) Reliability: 0.99 3) Power: 26 watts 4) Weight: 45 lb Volume: 2560 cu. in. 5) 1300 Integrated Circuits 2800 Discrete Parts 2 Memories 6) Equivalent to Configuration No. 1.	1) Meets the requirements. 2) Reliability: 0.9917 3) Power: 104 to 208 watts. 4) Weight: 104 to 208 lb. Volume: 5100 to 10,200 cu. in. 5) 2800 to 5600 Integrated Circuits 700 Discrete Parts 4 Memories 6) Implementation is extremely complex	Meeting Criteria 3 & 5, "2"* Reliability 5, 1&4, 2,3 Power 4, 2&1, 3, 5 Wt. & Volume 1 & 2 & 3, 4, 5 Cost & Prts. Ct. 1 & 4, 2, 3, 5 Design Complex 1 & 3 & 4, 2, 5 Perform 2, 3 (4, 5, 1 not accept) SELECTED APPROACH No. 2 Dependent Parallel Synchronous DPU

Figure 1.2.8-7: DATA PROCESSOR CONFIGURATION TRADE STUDY

<table>
<tr><td>TRADE STUDY SUMMARY SHEET</td><td>SOURCE OF REQUIREMENT</td><td colspan="2">TRADE STUDY NUMBER & TITLE — MEMORY SUBASSEMBLY TYPE SELECTION</td><td rowspan="3">SELECTION</td></tr>
<tr><td colspan="2" rowspan="2">FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS</td><td colspan="2">MATRIX OF DESIGN APPROACH</td></tr>
<tr><td>1. DESTRUCT READOUT</td><td>2. NONDESTRUCT READOUT</td></tr>
<tr><td colspan="2">Storage Capacity Requirements
• 1024 words, 27 bits each.
• Random access by word.
• Sequential by bit.
• Bit time ≤ 100 μ secs
• Maximum power < 2 watts at 10 KHz average
• Weight ≤ 1.82 Kg
• Volume ≤ 10.16 cm by 19.81 cm by 15.87 cm
• Reilability ≥ 0.940

The parameters were considered in the following order of importance:
1. Affecting the criteria
2. Reliability
3. Power
4. Weight and volume
5. Cost or parts count
6. Design complexity
7. Performance</td><td>1) Meets criteria adequately.
2) Reliability > 0.940
3) Power < 2 watts average
4) Weight & Volume < meets spec.
5) Fewer parts lower cost
6) Simplest and space-proven
7) Requires bit regeneration.</td><td>1) Meets criteria adequately
2) Reliability - not available
3) Power < 2 watts average
4) Weight & Volume data not confirmed but expected to meet requirements
5) At least four times as many semiconductor components higher cost
6) Most complex low signal-to-noise figures and not space-proven.
7) Does not require bit regeneration.</td><td>Meeting Criteria
1 & 2
Reliability
1
Power
2, 1
Weight & Volume
1, 2
Parts
1, 2
Cost
1, 2
Complexity
1, 2
Performance
2, 1

SELECTED APPROACH
1</td></tr>
</table>

Figure 1.2.8-8: MEMORY TYPE TRADE STUDY

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE: RANDOM VERSUS SEQUENTIAL ACCESS COMPUTER MEMORY		SELECTION
		MATRIX OF DESIGN APPROACH		
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS		1. RANDOM ACCESS	2. SEQUENTIAL ACCESS	
Storage Capacity Requirements: • 1024 words, 27 bits each • Maximum power < 2 watts average at 10 KHz • Weight ≤ 1.82 Kg • Volume ≤ 10.16 cm by 19.81 cm by 15.87 cm • Reliability ≥ 0.940 The parameters were considered in the following order of importance: 1. Meeting criteria 2. Reliability 3. Power 4. Weight and volume 5. Cost and parts count 6. Design complexity 7. Performance		1. Meets criteria 2. Reliability > 0.940 3. Power < 2 watts average 4. Weight & volume - meets spec. 5. Parts count comparable but with less stringent tolerances. Cost 50% less. 6. Simplest and space-proven. 7. Lower operating speed, less power, inherently more reliable.	1. Performs storage function but does not meet physical requirements. 2. Reliability > 0.940 but not confirmed. 3. Power > 6 watts 4. Weight & volume - meets spec. 5. Parts count comparable but high speed operation requires strongest tolerances. 6. More complex but space-proven. 7. High operating speed, more power lower signal-to-noise, EMI sensitivity potentially less reliable.	1. Meeting Criteria 1, 2 2. Reliability 1, 2 3. Power 1, 2 4. Weight & Volume 1, 2 5. Parts & Cost 1, 2 6. Complexity 1, 2 7. Performance 1, 2 SELECTED APPROACH 1

Figure 1.2.8-9: COMPUTER MEMORY ACCESS TRADE STUDY

1.2.9 Propulsion Subsystem

1.2.9 Propulsion Subsystem

The Task D propulsion subsystem is an all liquid propellant system utilizing the propellants monomethylhydrazine ($CH_3N_2H_3$) and nitrogen tetroxide (N_2O_4). The lunar module descent engine (LMDE) is used at 9850 pounds thrust to provide the necessary thrust for Mars orbit insertion and at 1050 pounds for all midcourse trajectory corrections and orbit trim maneuvers.

The propellant storage systems have been sized to provide a ΔV capability of 1.95 km/sec for the following 1979 Mars mission requirements:

Flight Capsule (lb)	Science Payload (lb)
7000	1056

A ΔV capability of 1.95 km/sec for the 1973 through 1977 Mars missions is provided without subsystem change by offloading propellants to accommodate lower planetary vehicle, flight capsule, and orbital science weights. Propellant requirements and target planetary vehicle weights for the 1973 through 1979 missions are shown in Figures 1.2.9-3 and 1.2.9-4.

1.2.9.1 Design Constraints and Requirements

The primary design requirements imposed upon the propulsion subsystem were:

1) Provide a single propulsion subsystem for all 1973 through 1979 Mars missions.

2) Use the lunar module descent engine as the prime propulsion system.

3) Consider the Titan III transtage engine and Agena Model 8517 engine as modular replacement engines.

4) Provide a minimum ΔV capability of 1.95 km/sec. for all missions.

5) Provide for a minimum of one arrival time biasing, two midcourse trajectory corrections, one orbit insertion, and two orbit trim maneuvers. The requirements for each of these maneuvers for the 1973 and 1979 Mars missions were established as follows:

	Midcourse & Arrival Bias (m/sec)	Orbit Insertion (m/sec)	Orbit Trim (m/sec)
1973	210	1590	150
1975	75	1775	100
1977	75	1775	100
1979	110	1740	100

6) Provide for the following payload capacities:

Flight Capsule (lb)	6000	7000	7000	7000
Science Payload (lb)	371	592	1001	1056

Additional design constraints placed upon the propulsion system were:

1) The propellant tankage is to be designed for the 1979 mission requirements. Propellant requirements for the 1973 through 1977 missions to be provided by off-loading the propellant tanks where necessary.

2) Propulsion subsystem design temperature limits were established at 5°C to 43°C.

3) The propulsion subsystem is to be designed for clean-room assembly. Non-self-sterilizing propellants and pressurants to be filtered and loaded aseptically.

1.2.9.2 Functional Description and Performance Characteristics

The propulsion subsystem is designed to provide the necessary impulse to accomplish the various velocity change maneuvers as required by the Voyager 1973 through 1979 Mars missions. As specified in the design requirements, a total of six operations are required. These operations include three maneuvers before orbit insertion for arrival time biasing of the two planetary vehicles and midcourse trajectory corrections, one orbit insertion maneuver, and two adjustment maneuvers in Mars orbit.

The propulsion subsystem selected to accomplish all required maneuvers is a multiple restart liquid bipropellant rocket system.

The propellants are monomethylhydrazine ($CH_3N_2H_3$) and nitrogen tetroxide (N_2O_4). Thrust is provided by a single lunar module descent engine operating at 1050 or 9850 pounds thrust. High pressure helium expanding through pressure regulators provides the means for propellant expulsion. A functional block diagram of the system is shown in Figure 1.2.9-1.

The system is shown schematically in Figure 1.2.9-2. The LMDE operating at 1050 pounds thrust is fed from the positive expulsion tanks, SO and SF, for a sufficient period of time to settle the propellants in the main propellant tanks, MO and MF, in the low gravity environment of space. The positive expulsion propellant system is also used for ΔV maneuvers of less than 10 m/sec. Propellants from the main tanks, MO and MF, are fed to the LMDE for all large ΔV maneuvers.

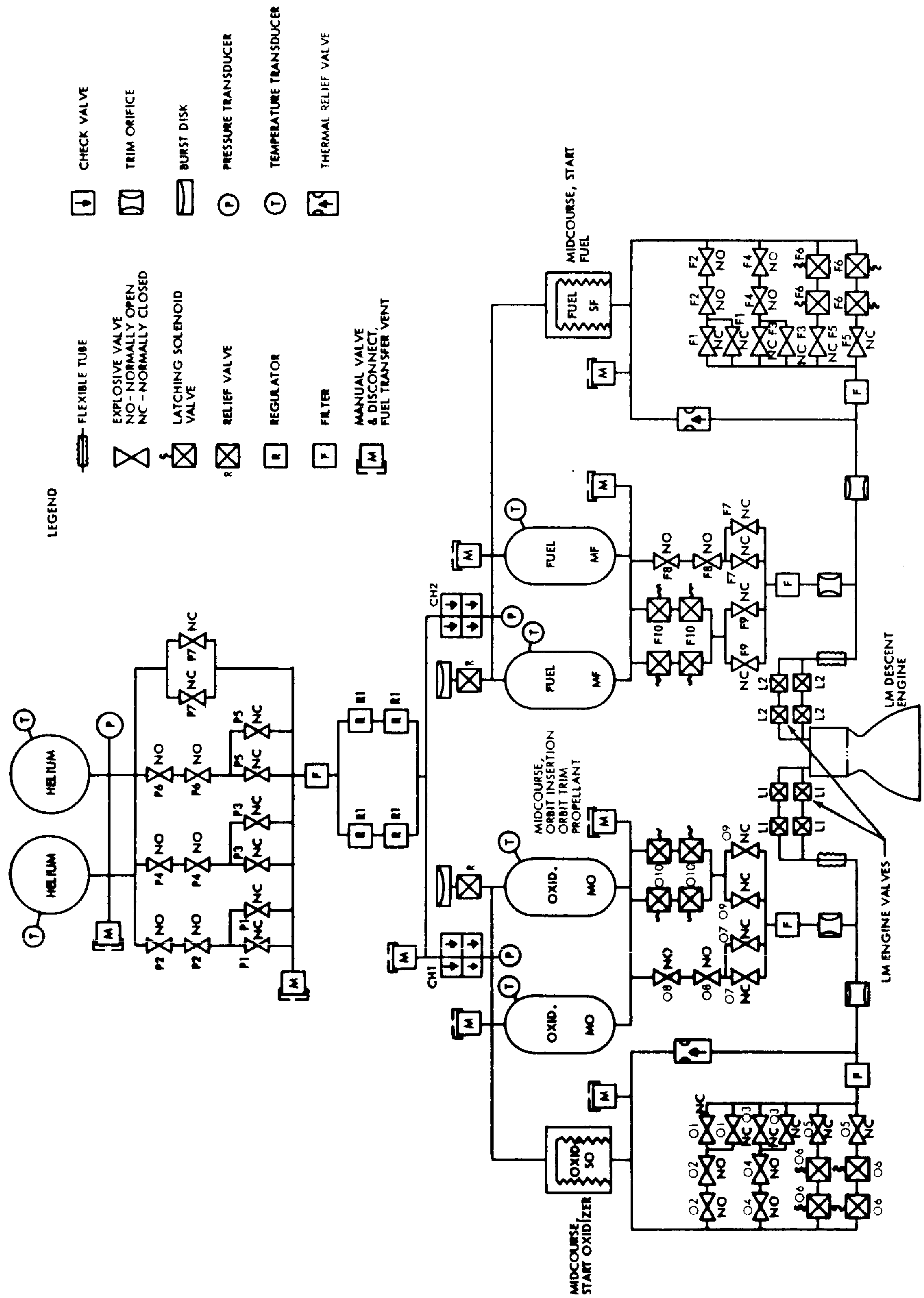


Figure 1.2.9-2 PROPULSION SUBSYSTEM SCHEMATIC

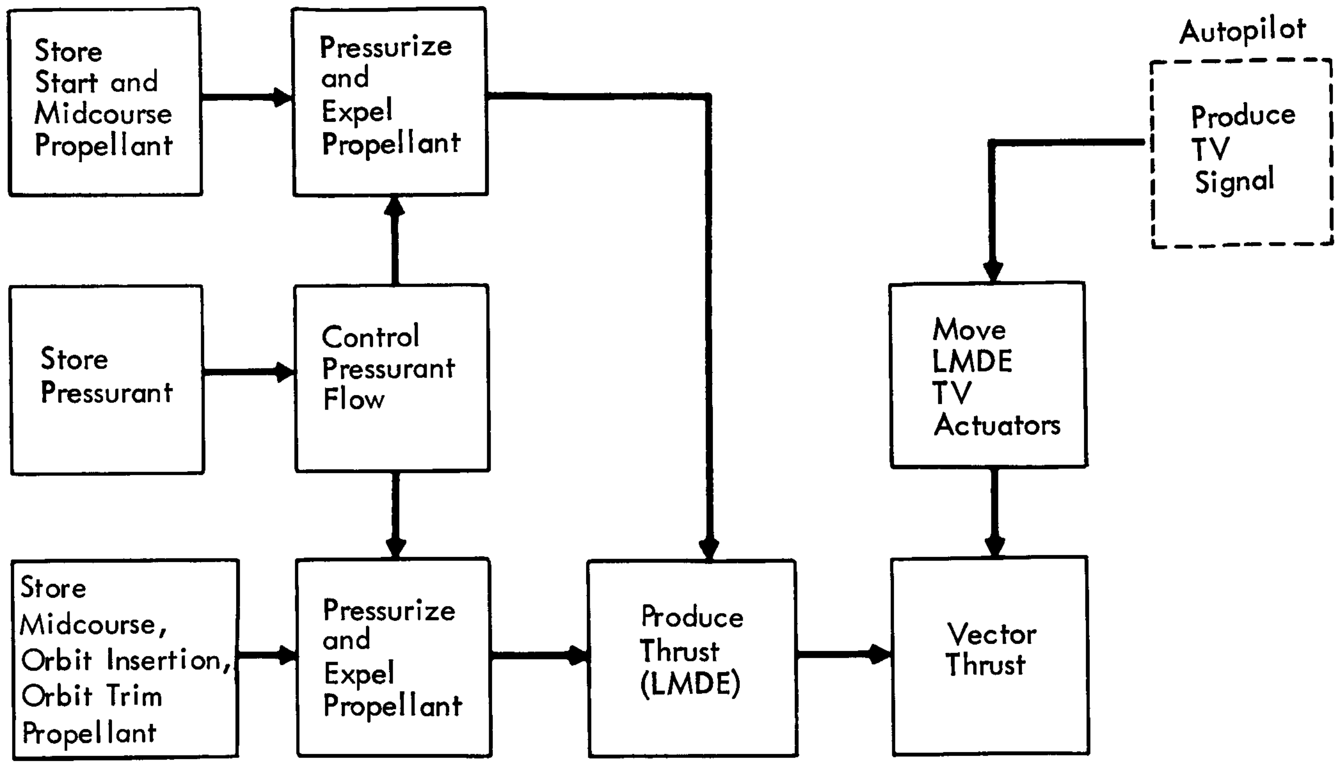


Figure 1.2.9-1: PROPULSION SUBSYSTEM FUNCTIONAL BLOCK DIAGRAM

A typical mission operational sequence is as follows:

1) First midcourse and arrival bias maneuver:

Positive isolation valves, P1, are fired open allowing helium to flow through regulators, R1, and check valves, CH1, and CH2, and pressurize the start propellant tanks, SO and SF, and the main propellant tanks, MO and MF.

Isolation valves, O1 and F1, are fired open allowing propellant flow from the start tanks to the engine. The engine valves, L1 and L2, are then opened initiating engine operation.

After approximately 7 seconds of engine operation, the main tank propellants are settled and the isolation valves, O7 and F7, are fired open allowing main tank propellants to flow to the LMDE inlet.

Approximately 2 seconds after LMDE ignition, isolation valves, O2 and F2, are fired closed isolating the start propellant tanks.

At the end of the midcourse maneuver, the LMDE inlet valves are closed shutting down the LMDE. The isolation valves, O8 and F8, are then fired closed followed by firing the isolation valves, P2, closed, isolating the main propellant tanks and the helium pressurant storage from leakage until the next maneuver.

2) Second and third midcourse maneuvers:

The sequence described above for operation of the start propellant tank system is repeated using isolation valves P3 and P4, O3 and O4, with F3 and F4, for the second midcourse maneuver and isolation valves P5 and P6, O5 and F5, plus the solenoid valves, O6 and F6, for the third midcourse. The main propellant tank feed system is not opened during these maneuvers as the total ΔV required is less than 10 m/sec. The start propellant tanks are isolated by means of the quad-redundant solenoid valves, O6 and F6, following the third midcourse maneuver, since the storage time until orbit insertion is considered short enough such that solenoid valve leakage can be tolerated.

3) Orbit insertion maneuver:

Positive isolation valves, P7, are fired open allowing the start propellant tanks and the main propellant tanks to be pressurized with helium. Solenoid valves, O6 and F6, are opened allowing propellant from the start tanks to flow to the engine inlets. The engine inlet valves, L1 and L2, are then opened and LMDE ignition occurs.

Approximately 16 seconds after engine ignition the main tank propellants are settled and isolation valves, O9 and F9, are fired open followed by opening the solenoid valves, O10 and F10, allowing propellant from the main propellant tanks to flow to the LMDE.

Two seconds following LMDE ignition the solenoid valves, O6 and F6, are closed shutting down the start propellant system. The LMDE is then commanded to the high thrust position.

System operation is terminated at the end of the maneuver by closing the LMDE inlet valves, L1 and L2, shutting down the LMDE. Solenoid valves O10 and F10 are then closed isolating the main propellant tanks.

4) Orbit trim maneuvers

The maneuvers are initiated by opening the start propellant tank solenoid valves and the LMDE inlet valves, thereby igniting the engine.

After 5 seconds of engine operation to settle the main tank propellants, the main propellant tank solenoid valves are opened to supply propellant to the LMDE. Two seconds later, the start tank propellant feed is shut down by closing the start tank feedline solenoid valves.

The maneuver is terminated by closing the LMDE inlet valves, L1 and L2, shutting down the engine. This is followed by closing the main tank feedline solenoid valves.

Performance characteristics of the rocket engine and the propulsion subsystem are given in Table 1.2.9-1. The total impulse capability shown in Table 1.2.9-1 results from the 1979 Mars mission design requirement weights shown in Figure 1.2.9-3.

Table 1.2.9-1: PROPULSION SUBSYSTEM PERFORMANCE

ENGINE (Modified Lunar Module Descent Engine Using Monomethyl Hydrazine/Nitrogen Tetroxide Propellant)

Engine Characteristics	Thrust (lbf)	Nominal I_s - (lbf-sec)/(lbm)
Midcourse, Orbit Trim, Orbit Insertion Start	1050	288
Orbit Insertion	9850	304
Nominal Mixture Ratio		1.6
Expansion Ratio		47.5
Minimum Impulse Bit, Low Thrust		1225 lb-sec
Maximum Gimbal Angle		± 6 degrees

PROPULSION SYSTEM

	Total Impulse (lbf-sec)	Burn Time (seconds)	Propellant Used (lbm)
1979 Design Point			
Midcourse	287,000	273	998
Orbit Insertion	3,386,000	344	11138
Orbit Trim	141,000	133	489
TOTAL	3,814,000	750	12625
1973 Design Point			
Midcourse	465,000	443	1615
Orbit Insertion	2,630,000	267	8640
Orbit Trim	183,000	174	635
TOTAL	3,278,000	884	10890

Reduced orbital science and flight capsule payload requirements for the 1973-1977 Mars missions can be accommodated by offloading propellant for these missions. The total payload capability as a function of offloaded propellant weight for the design point propulsion subsystem is shown in Figure 1.2.9-4.

1.2.9.3 Physical Characteristics

The propulsion subsystem configuration is shown in Figure 1.1.7-3. The subsystem has been integrated into a module that can be assembled and tested or removed as a unit from the spacecraft. The main propellants are contained in four equal-volume spherical tanks, two fuel and two oxidizer. The propellant tank capability is designed to meet the 1979 Mars mission requirements, resulting in the following tank size:

Flight Capsule Weight (lb)	Science Payload Weight (lb)	Usable Propellant Weight (lb)	Tank Diameter (in.)
7000	1056	12325	54

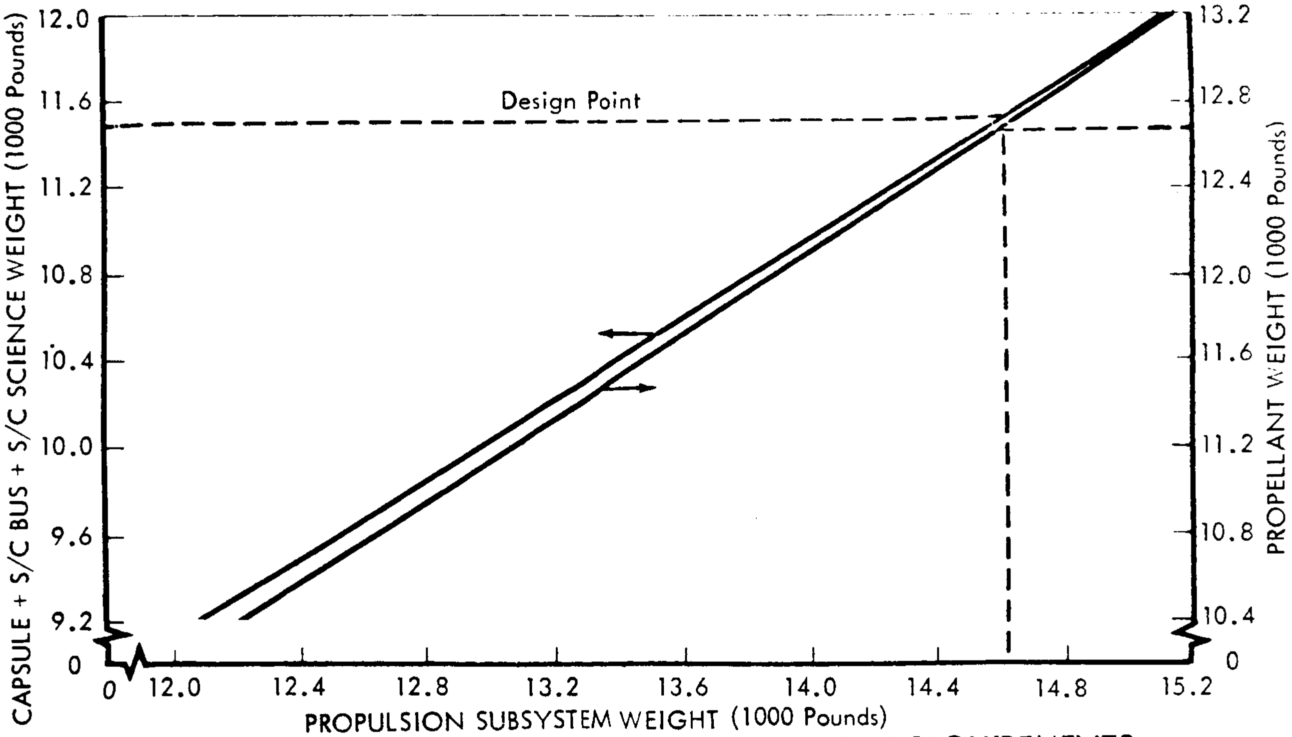


Figure 1.2.9-3: PROPULSION SUBSYSTEM SIZE REQUIREMENTS 1979 MARS MISSION

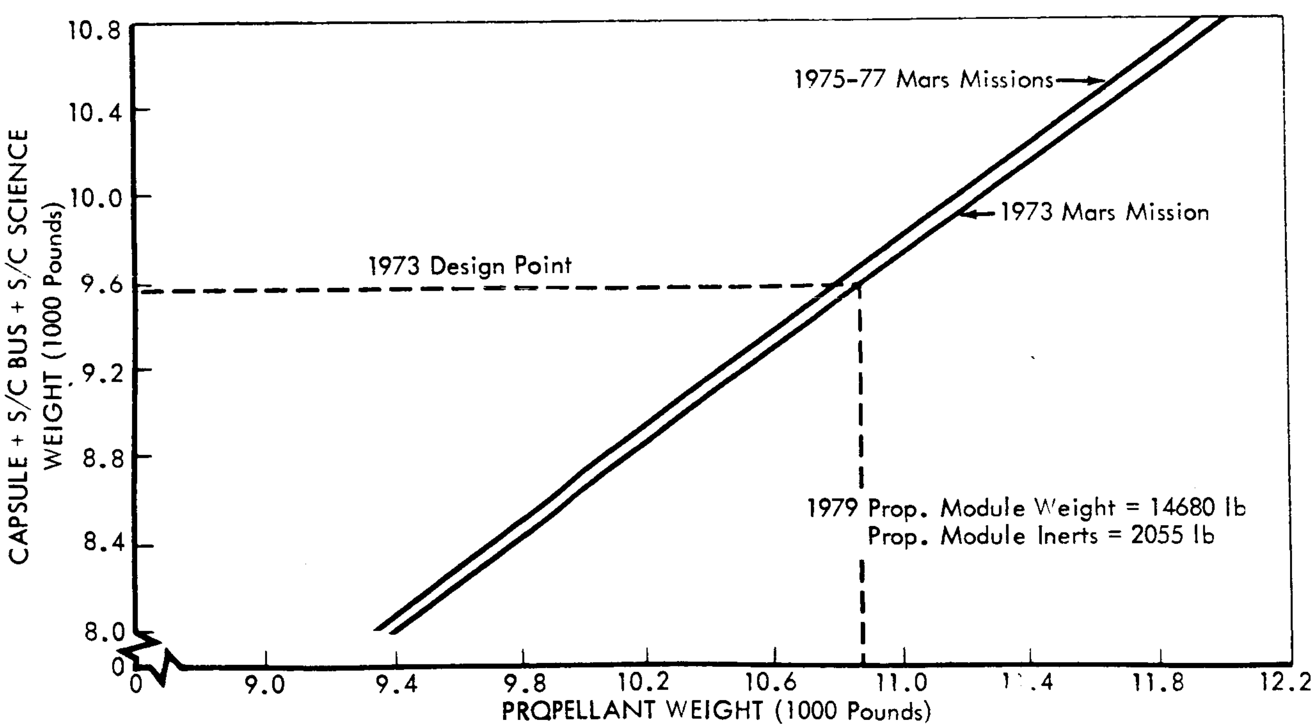


Figure 1.2.9-4: PROPULSION SUBSYSTEM CAPABILITY 1973-1977 MARS MISSION

In addition to the main propellant tanks, two positive expulsion bellows tanks containing 300 pounds of usable propellants are included to provide a controllable volume of propellant for main tank propellant settling under low gravity conditions plus propellant for sufficient impulse to produce 10 m/sec ΔV capability. These tanks are 20 inches in diameter by 32 inches long as shown in Figure 1.2.9-5.

The propellant is contained within a 16.75 inch diameter stainless steel bellows and pressurized from the helium pressurant supply as shown in the system schematic, Figure 1.2.9-2.

The lunar module descent engine is modified by removing the throttle actuator and venturi flow control valves used on Apollo. The movable injector element is moved from the low thrust position to the maximum thrust position by means of a hydraulic actuator using fuel pressure actuation. The engine is supported by means of its gimbal mounts on the centerline at the rear of the propulsion module thrust structure. The engine exhaust plane is positioned 86 inches aft of the solar panels and the columbium nozzle extension is covered with external insulation to prevent overheating of the solar cells. Thrust vector control is provided by gimballing the engine in the pitch and yaw planes by means of redundant electromechanical actuators. Roll torques produced by engine firing are controlled by the roll control nozzles of the guidance and control subsystem reaction control system. Engine configuration and overall dimensions are given in Figure 1.2.9-6.

The propulsion subsystem plumbing is designed to prevent mission failure from a single component failure and to minimize propellant or pressurant leakage during long storage periods in space. Explosive squib valves in redundant pairs are used to isolate the helium storage system before and after each midcourse maneuver. Following the orbit insertion maneuver, the storage time between maneuvers is less than 10 days, and nominal leakage rates can be tolerated. Although positive isolation valves are not used in this time period, quad-redundant regulators in the helium feed lines will prevent loss of helium through the propellant tank relief valve in the event of single regulator abnormal leakage.

The propellant tanks are likewise isolated between midcourse maneuvers by redundant pairs of squib valves. Following the third midcourse maneuver, the start propellant tanks are isolated by quad-redundant solenoid valves. Since the storage time before the next maneuver (orbit insertion) is 30 days or less, nominal solenoid valve leakage can be tolerated. The start propellant tank shutoff valves are backed up during this period by the LMDE inlet valves, and therefore the propellant loss to space will be extremely small.

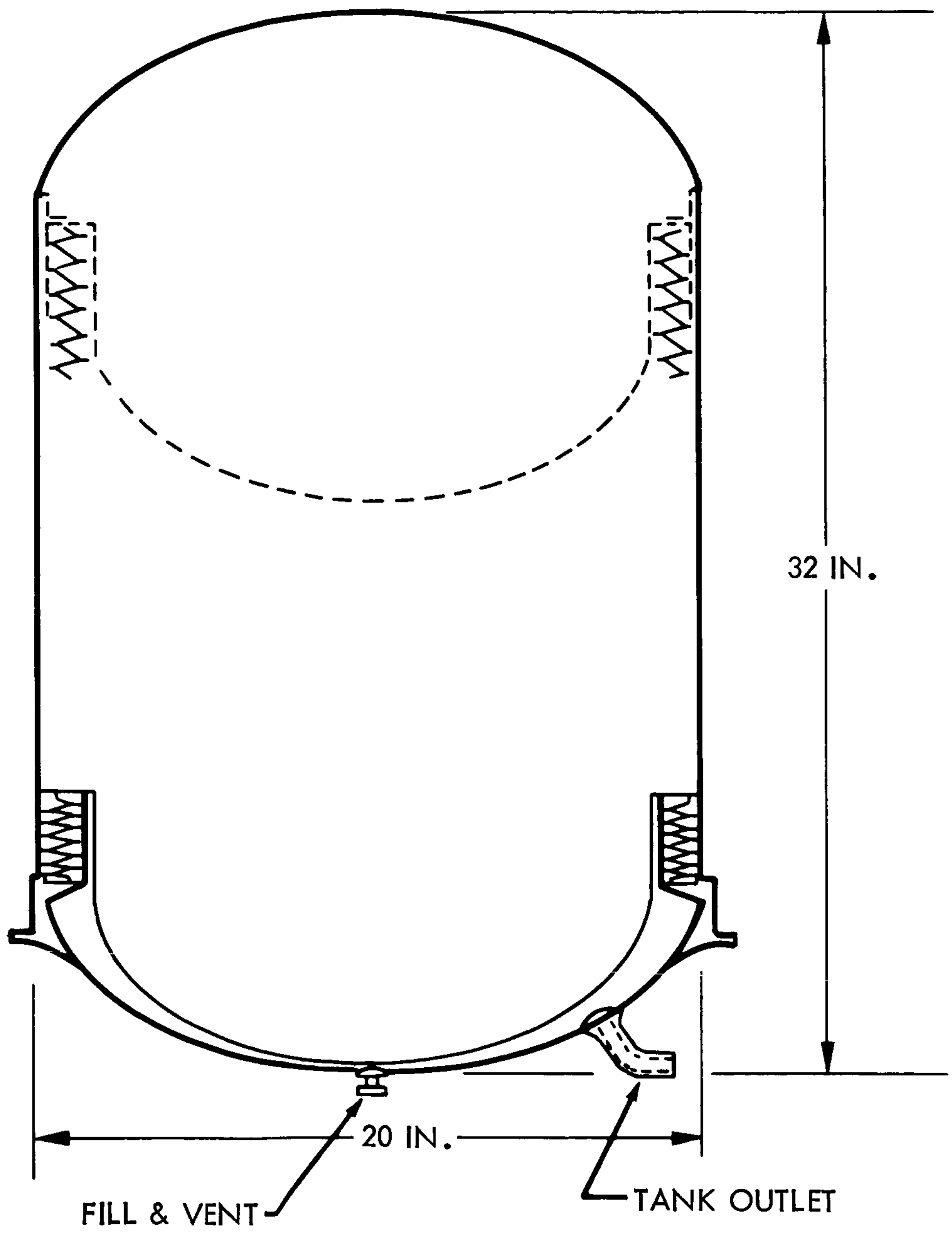


Figure 1.2.9-5: POSITIVE EXPULSION BELLOWS TANK

BLANK

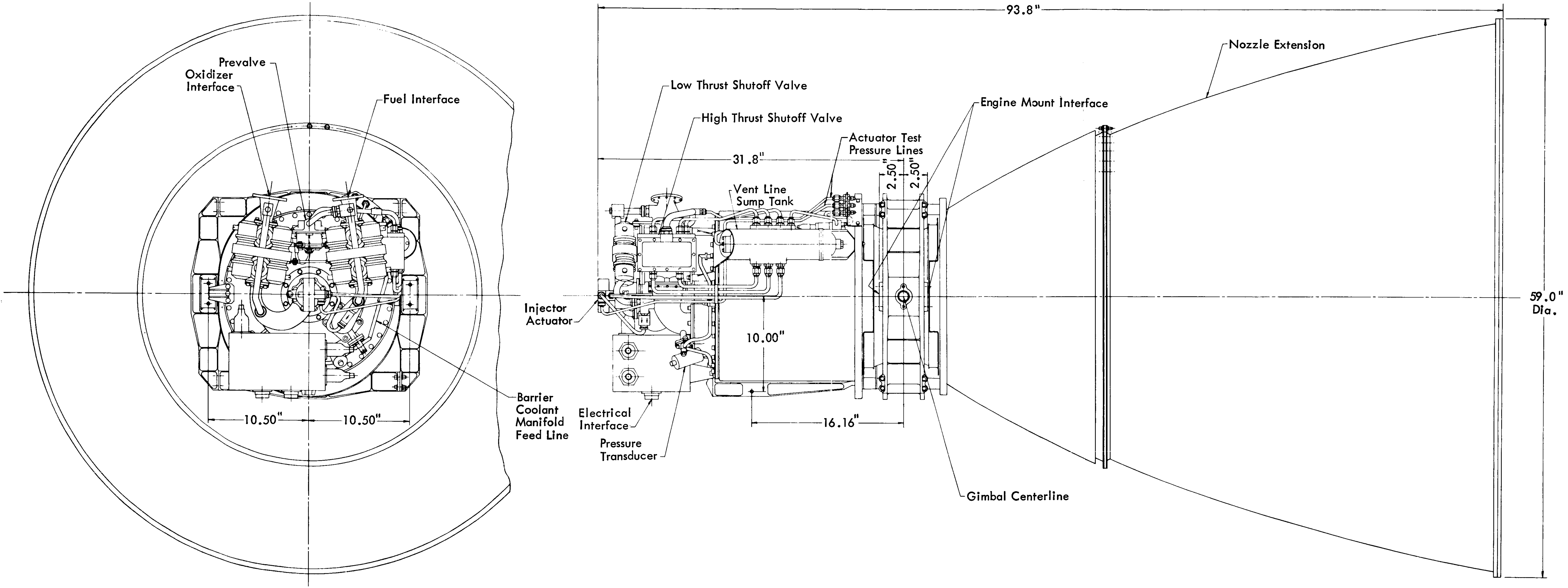


Figure 1.2.9-6: VOYAGER LUNAR MODULE DESCENT ENGINE

The pressure regulators in the helium supply lines are connected in a series-parallel quad-redundant arrangement to prevent loss of the mission from a single regulator failure. However, regulator set pressures are offset such that only a single regulator is controlling tank pressure at any time to prevent regulator hunting and damaging pressure surges. Assuming a set pressure tolerance of ±1% and lockup at +2%, regulator number 1 in a series leg will be set at a nominal working pressure of 245 ± 2 psia with lockup at 250 psia. Then regulator number 2 in the same series leg will be set at 253 ± 2 psia with lockup at 258 psia. Regulator number 1 in the parallel leg will be set at 238 ± 2 psia with lockup at 243 psia and regulator number 2 in this series leg will be set at 246 ± 2 psia locking up at 251 psia. Thus, the nominal tank operating pressure will be 245 psia, but may be a maximum of 255 psia in the event of a regulator-open failure. In the event of a regulator-closed failure, the minimum tank operating pressure may be 236 psia. Such tolerances are considered to be within an acceptable range for engine operation; however, these tolerances could be tightened by individual selection and bench testing during system assembly.

A complete weight breakdown of the propulsion subsystem is given in Table 1.2.9-2.

Table 1.2.9-2: PROPULSION SUBSYSTEM WEIGHT SUMMARY

	WEIGHT (Pounds)		
	1973		1979
DRY SYSTEM			
Lunar Module Descent Engine		409	409
Propellant System		895	895
Tanks, Main	543		
Tanks, Bellows	77		
Tank Support	120		
Plumbing	155		
Pressurization System		471	471
Tanks	440		
Tank Supports	[1]		
Plumbing	31		
Engine Support & Miscellaneous		24	24
TOTAL PROPULSION DRY		1799	1799
RESIDUALS		378	378
Propellants	338		
Pressurants	40		
USABLE PROPELLANT (1973)		10890	
TOTAL PROPULSION (1973)	13067		
USABLE PROPELLANT (1979)			12625
TOTAL PROPULSION (1979)			14802

[1] Included in propellant system tank support above.

1.2.9.4 Interface Definition

The propulsion subsystem interfaces with the power, guidance and control, structural and mechanical, and other subsystems are shown in Table 1.2.9-3.

Table 1.2.9-3: PROPULSION SUBSYSTEM INTERFACES

ITEM. NO.	TYPE OF INTERFACE	INTERFACE DESCRIPTION	INTERFACING SUBSYSTEM	INPUT TO	INPUT FROM	BOUNDARY DEFINITION
1.	Electrical	Electrical power – 50 v rms at 2.4 kHz	Power		X	Cable connector at propulsion subsystem
2.	Electrical	Electrical power – 22-32 v d.c.	Power		X	Cable connector at propulsion subsystem
3.	Electrical	Propulsion subsystem telemetry signals (30 measurements)	Telemetry	X		Cable connector at propulsion subsystem
4.	Electrical	Commands to propulsion subsystem (7 commands)	Command		X	Cable connector at propulsion subsystem
5.	Electrical	Commands to propulsion subsystem (7 commands)	Computing & Sequencing		X	Cable connector at propulsion subsystem
6.	Structural	Field joint between propulsion module and spacecraft. Eight-bolt connection on approximately 158-inch diameter.	Structural & Mechanical, Thermal Control	X		Eight structural fittings
7.	Electrical	Initiation of the EED's in the pressurant and propellant system.	Pyrotechnic		X	Cable connector at propulsion subsystem
8.	Structural	Lower thermal shield attachment to propulsion module structure	Propulsion Thermal Control		X	Insulation system attached to structure.
9.	Structural	Engine heat shield	Propulsion Thermal Control		X	Insulation system attached to structure.
10.	Mechanical	Propellant tank fill connection Pressure tank fill connection Propellant tank vent connection	OSE Fuel Servicing Unit	X	X	Mechanical plumbing connection to allow propellant and pressurant flow
11.	Mechanical	Propellant tank fill connection Pressure tank fill connection Propellant tank vent connection	OSE Propellant System Purge, Dry, and Flush Unit	X	X	Mechanical plumbing connection to allow purging, drying and flushing of system.
12.	Mechanical	Plumbing test connections in the pressurant and propellant plumbing	OSE Propulsion System Test Unit	X	X	Mechanical plumbing connections to allow testing of system.
13.	Electrical	Electrical connections to check wiring continuity, operation of valves, record system pressure and temperatures, and operate jet vanes	OSE Propulsion System Test Unit	X	X	Electrical connection to allow testing of system
14.	Mechanical	Catalac black thermal coating	Propulsion Thermal Control	X		Surface of tank side of solar shield and exterior of all tanks except motor case.
15.	Mechanical	Thrust vector actuators (2 places)	Guidance & Control	X		Attachment to motor case

1.2.9.5 Reliability

The propulsion subsystem was designed to meet the reliability goals established to ensure a high probability of mission success. The approach was to use redundant components where possible to eliminate mission loss from a single component failure. In addition, isolation of propellants and helium pressurant during long storage periods in space was provided to minimize the possibility of leakage losses due to unpredictable valve leakage after operation in space.

The reliability goal established for the propulsion subsystem was 0.995 for the 1973 mission and the system reliability is shown in Table 1.2.9-4.

Table 1.2.9-4: PROPULSION SUBSYSTEM RELIABILITY

ITEM	VALUE	ITEM	VALUE
Tank, Helium	0.99902	Valve, Squib NO, Prop., Start	0.999999
Valve, Squib NC, Helium	0.999999	Valve, Solenoid, Start	0.999985
Valve, Squib NO, Helium	0.999999	Filter, Propellant, Start	0.999989
Filter, Helium	0.999994	Orifice, Start	0.999999
Regulator	0.999954	Valve, Squib NC, Prop.	0.999999
Check Valve	0.999968	Valve, Squib NO, Prop.	0.999999
Burst Disk and Relief Valve	0.999999	Filter, Propellant	0.999999
Tubing Assemblies, Helium	0.99976	Orifice	0.999999
Bellows, Propellant	0.999934	Engine, LMDE	0.999917
Tank, Propellant	0.9981	Tubing Assemblies, Prop.	0.99984
Valve, Squib NC, Prop., Start	0.999999		

Subsystem Reliability_____0.9965

The system reliability exceeds the reliability goal. An inspection of Table 1.2.9-4 shows that, due to the plumbing system redundancy, the propellant and pressurant tanks are the main contributors to the system reliability degradation. Failure rates attributable to the tankage are based on normal aerospace industry inspection and leakage detection. Evidence indicates that improved quality control methods can reduce tankage failure rates to 50% of those used to determine the system failure rate shown in the table. Thus, by instituting improved inspection and quality control procedures during the manufacture and testing of the Voyager propulsion subsystem tankage, the system can be made to exceed its reliability goal by a considerable margin.

An examination of the trade study summary, Figure 1.2.9-7, shows that the inclusion of the C-1 engines in the propulsion subsystem degrades the subsystem reliability when compared to a propulsion subsystem using the LMDE for all maneuvers. However, the inclusion of the C-1 engines allows the potential use of these engines as a backup mode for orbit insertion if the LMDE fails to ignite. This potential could enhance probability of mission success even though the system reliability in a normal operating mode is not increased, and makes this system worthy of considering as an alternate.

1.2.9.6 Trade Study Summary

Propulsion subsystem trade studies were conducted to compare the use of the LMDE and the LMDE/C-1 combination, and to compare the modular interchangeability of the LMDE with the Transtage and Agena Model 8517. The results of these trade studies are shown in the following Trade Study Summary Sheets, Figures 1.2.9-7, 1.2.9-8, and 1.2.9-9.

1.2.9.7 New Technology and Development Items

No new technology or development is required for the propulsion subsystem.

1.2.9.8 Growth Potential

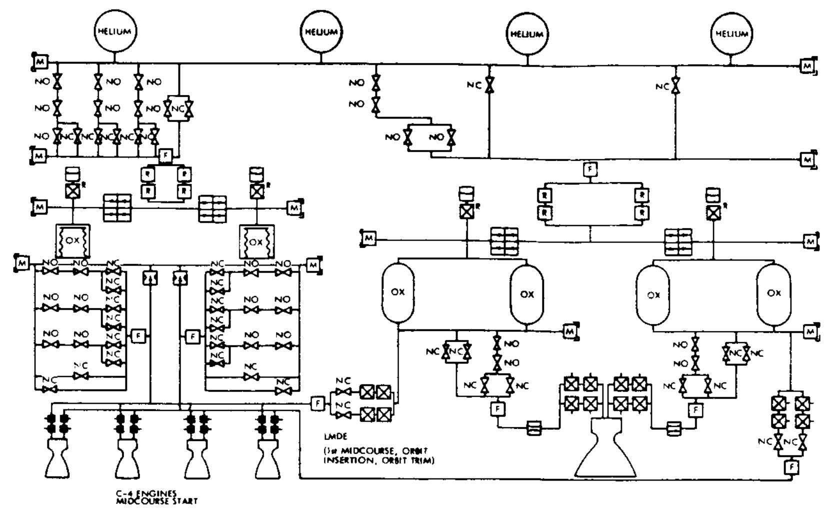
The propulsion subsystem defined for the 1973 baseline spacecraft is sized for the 1973 to 1979 missions. The growth required for the 1975-1979 missions is provided by increasing the amount of usable propellants loaded.

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE	SELECTION
		PROPULSION SUBSYSTEM SIZING — CONCEPT SELECTION	

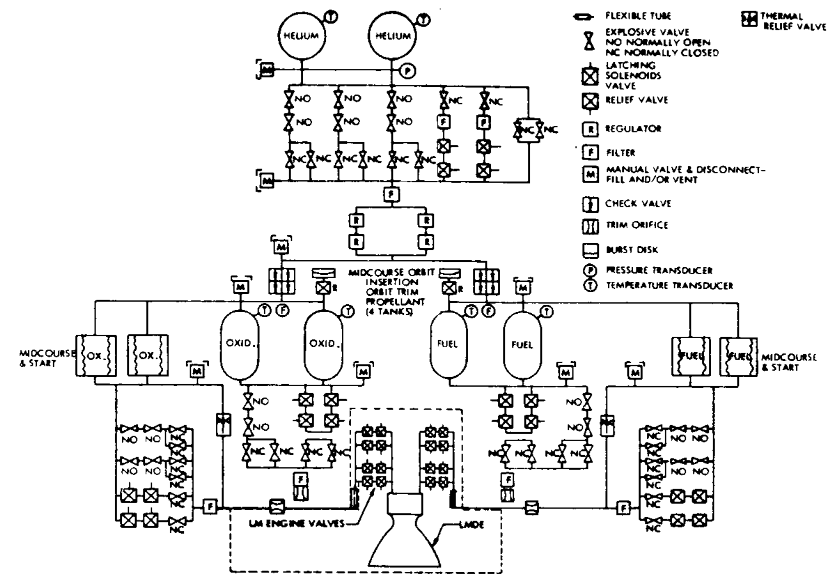
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

1. Provide a single propulsion subsystem for all 1973 through 1979 Mars missions.
   a. Propellant tankage to be designed for 1979 Mars mission requirements.
   b. Provide a minimum ΔV capability of 1.95 km/sec for all missions.
2. Provide for a minimum of one arrival time biasing, two midcourse trajectory corrections, one orbit insertion, and two orbit trim maneuvers.
3. Use of the Lunar Module Descent Engine as the prime propulsion system.
4. The plumbing system to be designed such that no single component failure will result in mission failure.

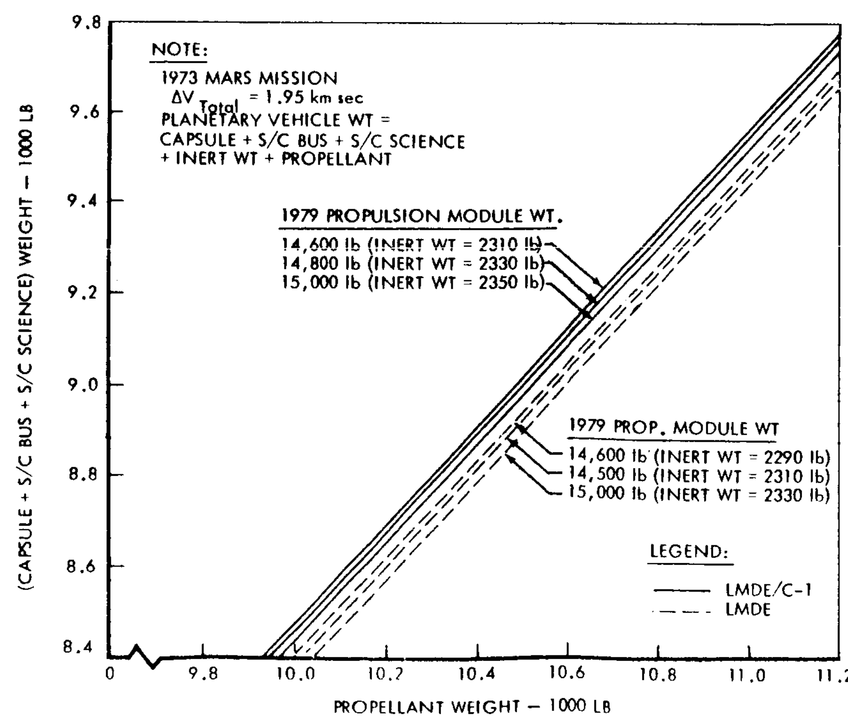
MATRIX OF DESIGN APPROACH



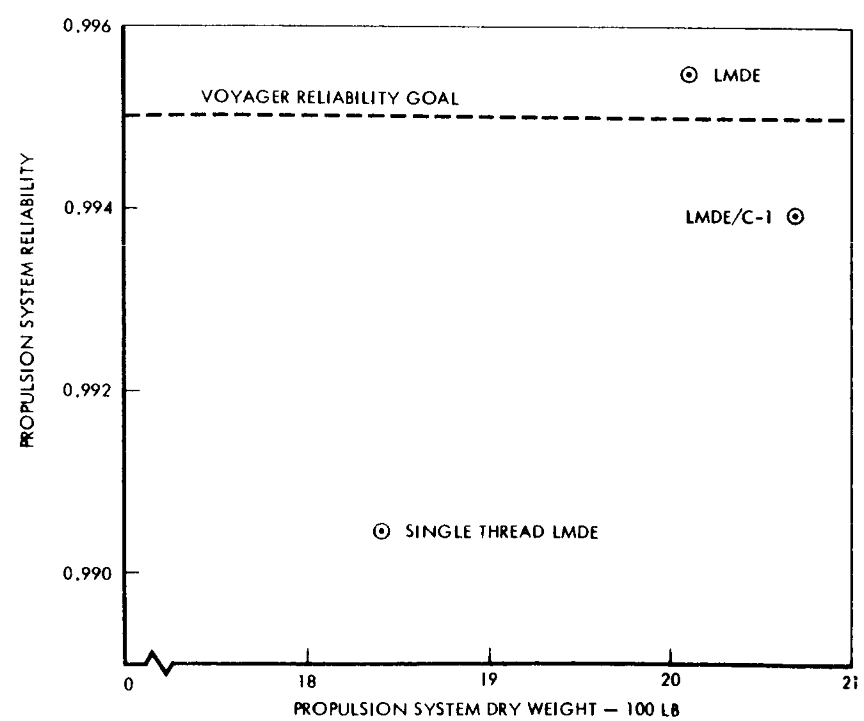
LMDE/C-1



LMDE



1973 MISSION CAPABILITY



RELIABILITY COMPARISON

SELECTION

1. LMDE/C-1 shows 60 lb 1973 payload advantage.
2. LMDE/C-1 has potential backup orbit insertion mode.
3. LMDE meets reliability goal

System selection cannot be made on basis of performance and reliability.

SELECTED APPROACH

LMDE selected to simplify the guidance and control subsystem.

Figure 1.2.9-7: PROPULSION SUBSYSTEM SIZING TRADE STUDY

TRADE STUDY SUMMARY SHEET

SOURCE OF REQUIREMENT

TRADE STUDY NUMBER & TITLE — ENGINE INTERCHANGEABILITY

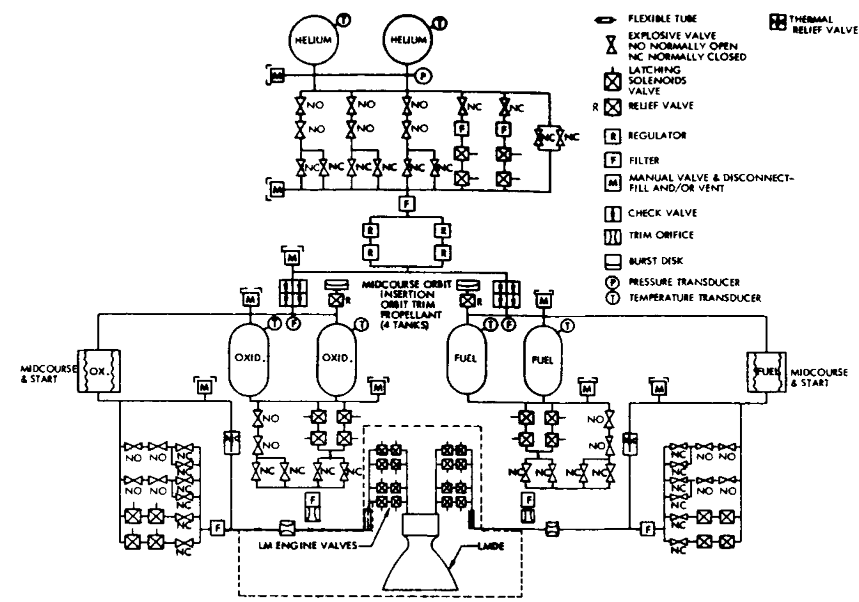
MATRIX OF DESIGN APPROACH

SELECTION

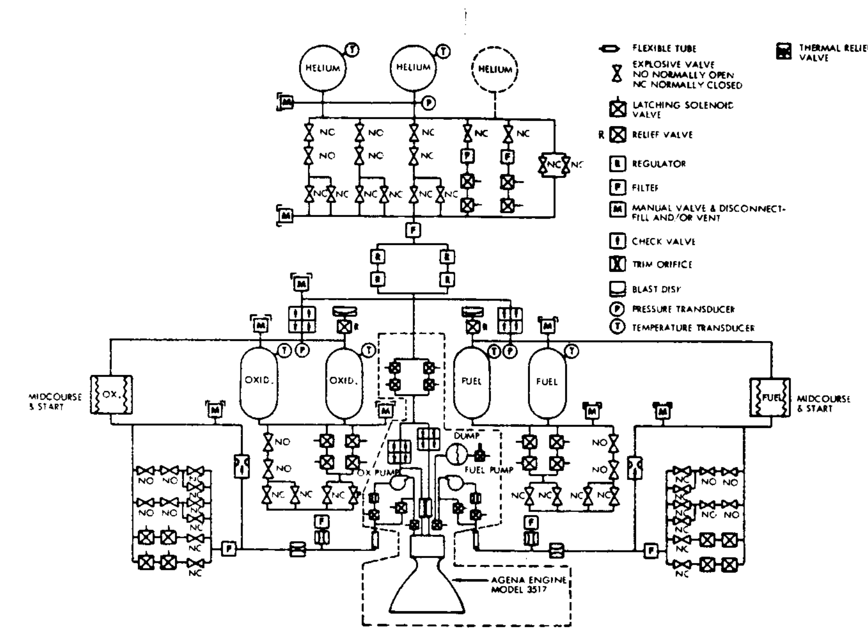
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

1. Provide a single propulsion subsystem for all 1973 through 1979 Mars missions
   a. Propellant tankage to be designed for 1979 Mars mission requirements
   b. Provide a minimum ΔV capability of 1.95 km/sec for all missions
2. Provide for a minimum of one arrival time biasing, two midcourse trajectory corrections, one orbit insertion, and two orbit trim maneuvers
3. Use of the Lunar Module Descent Engine as the prime propulsion system
4. Consider the use of the Titan III transtage engine and Agena Model 8517 as a modular replacement for the LMDE

MATRIX OF DESIGN APPROACH



LMDE



AGENA

SUBSYSTEM CHANGES REQUIRED FOR AGENA:

- Revise engine inlet plumbing to provide turbopump bypass for pressure-fed mode.
- Oriface pump inlets to reduce pump inlet pressure from 180 psia to 26 - 41 psia normal inlet pressure.
- Revise feed system plumbing to mate with Agena inlet geometry.
- Provide reservoir system or zero thrust overboard dump for fuel dumped during engine start and shutdown.
- Add helium tank and feed plumbing for low thrust Agena operation.
- Revise thrust structure to mate with Agena gimbal geometry.
- Provide thermal shield for turbine hot exhaust.

POTENTIAL PROBLEM AREAS FOR AGENA VOYAGER APPLICATION:

- Engine pump inlet static seals — redesign for higher pressure.
- Engine regenerative cooling system — redesign for low flow with helium saturated propellants.
- Engine injector — redesign for helium injection.
- Nozzle skirt extension – addition of insulation for thermal control.
- Engine performance – requalification for MMH and MR = 1.65.
- Thrust vector control — actuator design compatible with Agena geometry, inertias, thrust misalignment.
- Spacecraft design — structural redesign for higher thrust loads.
  — Thermal shielding for turbine exhaust.

SELECTION

1. LMDE plumbing system is less complex
2. Agena offers potential back-up mode
3. Agena engine as required for Voyager will not be space proven on other programs

SELECTED APPROACH

LMDE is preferred engine

Figure 1.2.9-8: ENGINE INTERCHANGEABILITY TRADE STUDY

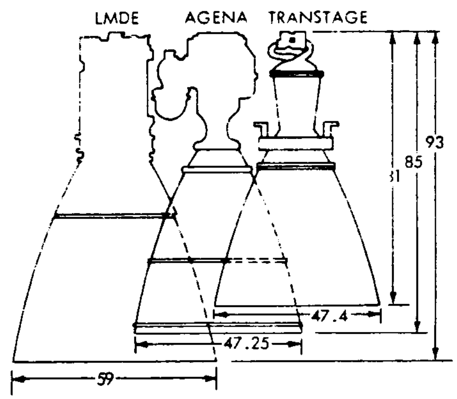
TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE: ENGINE INTERCHANGEABILITY	SELECTION
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS		MATRIX OF DESIGN APPROACH	

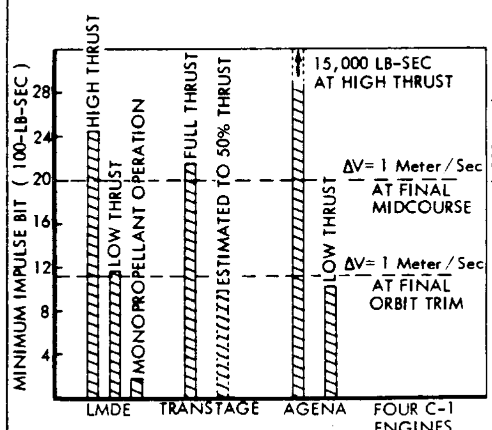
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

1. Provide a single propulsion subsystem for all 1973 through 1979 Mars missions.
   a. Propellant tankage to be designed for 1979 Mars mission requirements.
   b. Provide a minimum ΔV capability of 1.95 km/sec for all missions.
2. Provide for a minimum of one arrival time biasing, two midcourse trajectory corrections, one orbit insertion, and two orbit trim maneuvers.
3. Use of the Lunar Module Descent Engine as the prime propulsion system.
4. Consider the use of the Titan III transtage engine and Agena Model 8517 as a modular replacement for the LMDE.

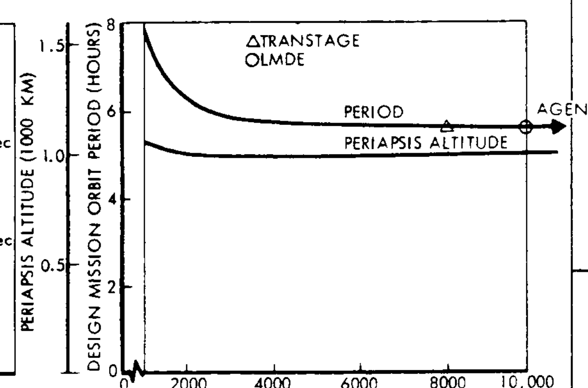
MATRIX OF DESIGN APPROACH

1. Envelope





MINIMUM IMPULSE BIT REQUIREMENTS



PRIMARY THRUST LEVEL (POUNDS)
EFFECT OF THRUST ON MISSION

2. Operating Characteristics
   a. Transtage
      MR = 2.0 - 7% Performance Penalty
      MR = 1.6 - 2% Performance Improvement
      Thrust - 80% of LMDE high thrust, no low thrust mode
      Inlet Pressure - Similar to LMDE, new trim orifices required
      TVC Range - 30% greater than LMDE
      Weight - 10% improvement
   b. Agena
      MR = 1.65 - similar to LMDE
      Thrust - 80% high, affects thrust structure
      Inlet Pressure - requires orificing pump inlets
      TVC Range - 40% less than LMDE, will meet Voyager requirements
      Weight - 12% Penalty for low thrust mode
      Thermal Control - additional thermal shield required around turbine, exhaust duct

SELECTION

SELECTED APPROACH

Transtage and Agena are interchangeable with LMDE with minor modifications.

Figure 1.2.9-9: ENGINE INTERCHANGEABILITY TRADE STUDY

BLANK

1.2.10 Structural and Mechanical Subsystem

1.2.10 Structural and Mechanical Subsystem

1.2.10.1 Design Constraints and Requirements

Structural and mechanical subsystem design constraints and requirements are summarized in Tables 1.2.10-1 and 1.2.10-2, respectively.

1.2.10.2 Functional Description and Performance Characteristics

Structural -- The spacecraft bus structural subsystem (Figure 1.2.10-2) provides structural support for all elements of the planetary vehicle. The structural system consists of primary load carrying trusses supporting the propulsion subsystem, the capsule, and the bus. The equipment bay assembly consists of stiffened skin with cable tray frames. Secondary structural items include antennas, solar panels, and support structure for antennas, platforms, and attitude control thrusters.

The primary truss is 2219 aluminum with welded end fittings. It is specifically sized to accommodate flight capsule weights up to 7000 pounds and propulsion system usable propellant weights up to 12,625 pounds. Critical design loads occur during boost, resulting in a combined limit load condition of 4.75-g axial and 2-g lateral. An ultimate factor of safety of 1.25 is provided. Propellant tankage is designed using annealed 6Al-4V titanium, and gas storage tankage is designed using 6Al-4V titanium heat treated to 160-ksi ultimate tensile strength. All pressure vessels are designed to an ultimate factor of safety of 2.0.

Eight electronic equipment modules (Figure 1.2.10-3) have 6061 aluminum thermal control radiator plates vertically stiffened at 12-inch intervals. Radiator plates double as shear webs for carrying equipment loading to the primary truss. The radiator plates vary in thickness from 0.05 to 0.25 inches.

Solar panels are constructed of 2024-T4 corrugation stiffened skins and frames bonded with high temperature epoxy (Epon 934 A/B) adhesive.

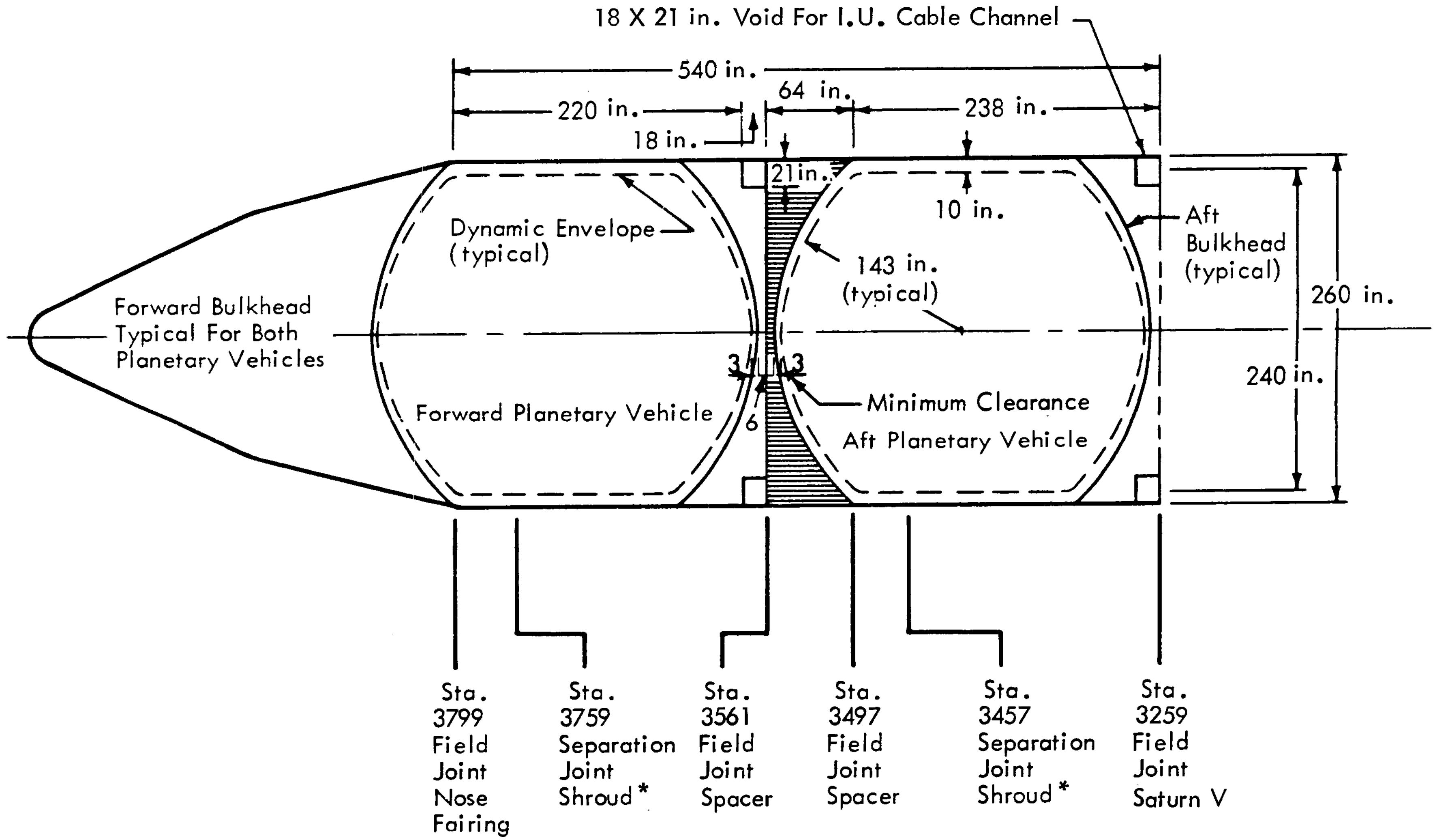
The high gain antenna is supported from a 6-inch diameter, 2219 boom with a minimum natural frequency of 2 cps.

No structure is provided specifically for the purpose of shielding for the meteoroid environment. Initial estimates of the added weight to the spacecraft for shielding are shown in Figure 1.2.10-4.

Table 1.2.10-1: STRUCTURAL DESIGN CONSTRAINTS AND REQUIREMENTS

DESIGN ENVIRONMENT	Natural per TM-X-53616 Induced per D2-82746-1
DESIGN LOAD FACTORS (ULTIMATE)	
Boost	
Singular, Torison	12.5 rad/sec^2
Combined	$N_X = +5.94$ & $N_{y,z} = \pm 2.5$
	or $N_X = -2.97$ & $N_{y,z} = \pm 2.5$
Orbit Insertion	$N_X = +1.10$ capsule on
	or $N_X = +2.10$ capsule off
DESIGN FACTORS OF SAFETY	
Structure	
Ultimate	1.25
Limit	1.00
Pressure Vessels	
Ultimate	2.0
Proof	1.5
Yield	1.65
MATERIAL ALLOWABLES	MIL-HDBK-5
RELIABILITY ALLOCATIONS	
Structure	0.999
ALLOCATED WEIGHTS (1973 Mission)	
Flight Capsule*	6000 lb
Flight Spacecraft (including science)	5800 lb
Spacecraft Science	390 lb
Flight Spacecraft Usable Propellant*	10,950 lb
DYNAMIC ENVELOPE	Figure 1.2.10-1

*Flight spacecraft structure designed to carry a 7000-pound flight capsule and the 1979 propulsion subsystem.



* Recommended for 8 point suspension support — Dependent on spacecraft configuration

Figure 1.2.10-1: SATURN V VOYAGER PLANETARY VEHICLE DYNAMIC ENVELOPE

Table 1.2.10-2: MECHANICAL DESIGN CONSTRAINTS AND REQUIREMENTS

FUNCTION	HIGH GAIN ANTENNA	MEDIUM GAIN ANTENNA	LOW GAIN ANTENNAS	SCIENCE SCAN PLATFORM	ULTRA VIOLET SPECTRO-METER	PLANETARY VEHICLE SEPARATION
Deployed Position Tolerance	$\pm 2.4^\circ$ Nom*	$\pm 2.4^\circ$ Nom	$\pm 2.4^\circ$ Nom	$\pm 2.4^\circ$ Nom	--	--
Launch Mode	Stowed	Stowed	Stowed	Stowed	--	Fastened
Deployment Period (Objective)	60 sec	60 sec	60 sec	60 sec	--	--
Stowed Hinge Moment (Minimum)	200 in-lb.	200 in-lb	200 in-lb	200 in-lb	--	--
Operating Mode	Deploy & latch	Deploy & latch	Deploy & latch	Deploy & latch	--	Release
Pointing Controls	2 Axis	1 Axis	--	3 Axis	2 Axis	--
Slew Rate ~ °/sec	2	2	--	2	2	--
Control Pulse Rate (Pulses per Second)	20	20	--	20	20	--
Pointing Accuracy (Nominal)	0.1°	1°	--	1°	1°	--
Rotation - Hinge Axis	$\pm 139^\circ$	$\pm 175^\circ$	--	$+160^\circ$ -15°	--	--
- Swivel Axis	$\pm 97^\circ$	--	--	$\pm 100^\circ$	$\pm 90^\circ$	--
- Azimuth Axis	--	--	--	$\pm 90^\circ$	$\pm 60^\circ$	--
Separation Weight	--	--	--	--	--	23,000 lb
Minimum Separation ΔV	--	--	--	--	--	1 ft/sec
Max Tipoff Rate °/sec	--	--	--	--	--	3
Rigidity & Damping (Deployed Position)	$\omega_n > 10$ rad/sec $\zeta \geq 0.01$	--	--	$\omega_n \geq 10$ rad/sec $\zeta \geq 0.01$	--	--

ω_n = Natural Frequency ζ = Damping

* $\pm 0.2^\circ$ Uncertainty after calibration

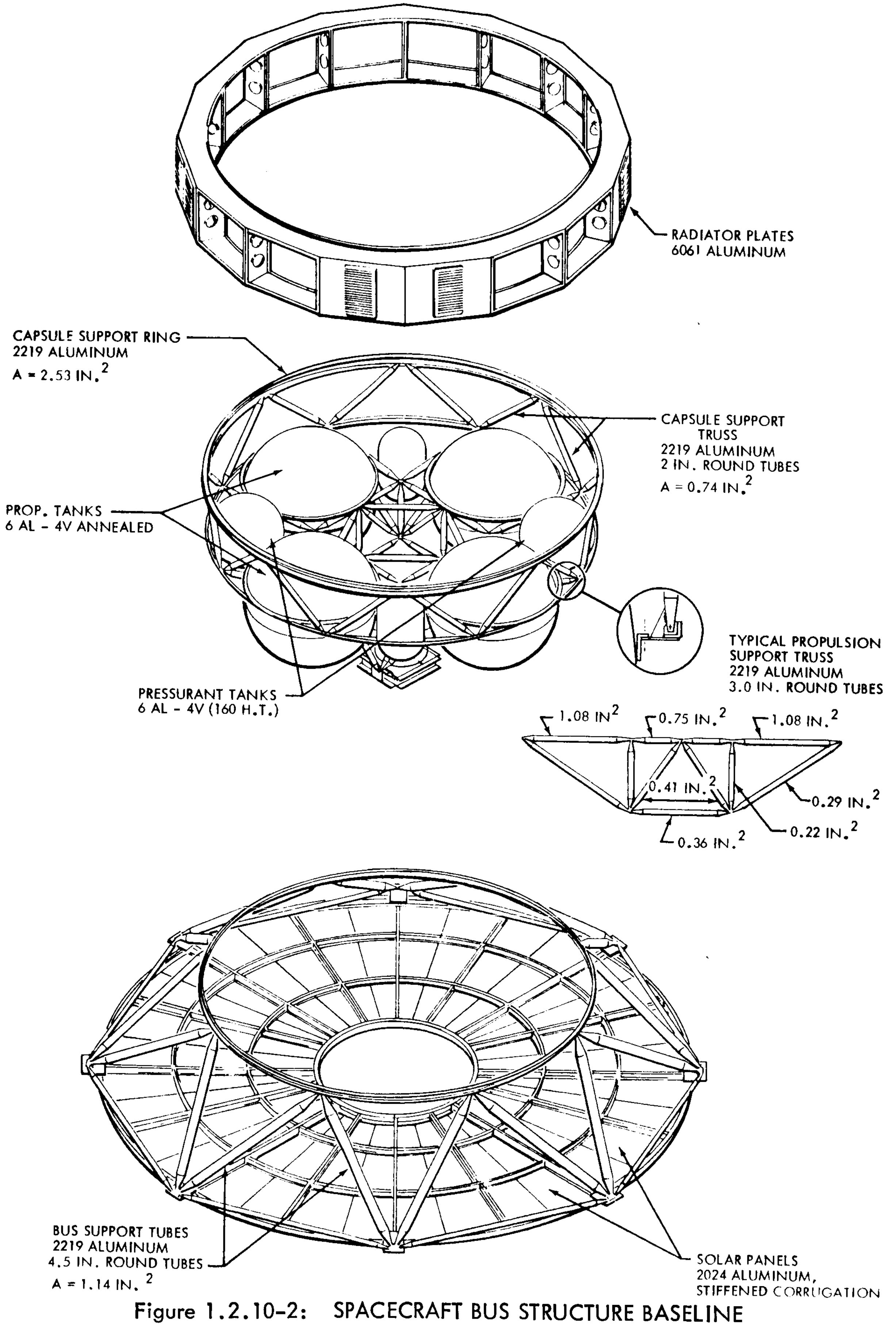


Figure 1.2.10-2: SPACECRAFT BUS STRUCTURE BASELINE

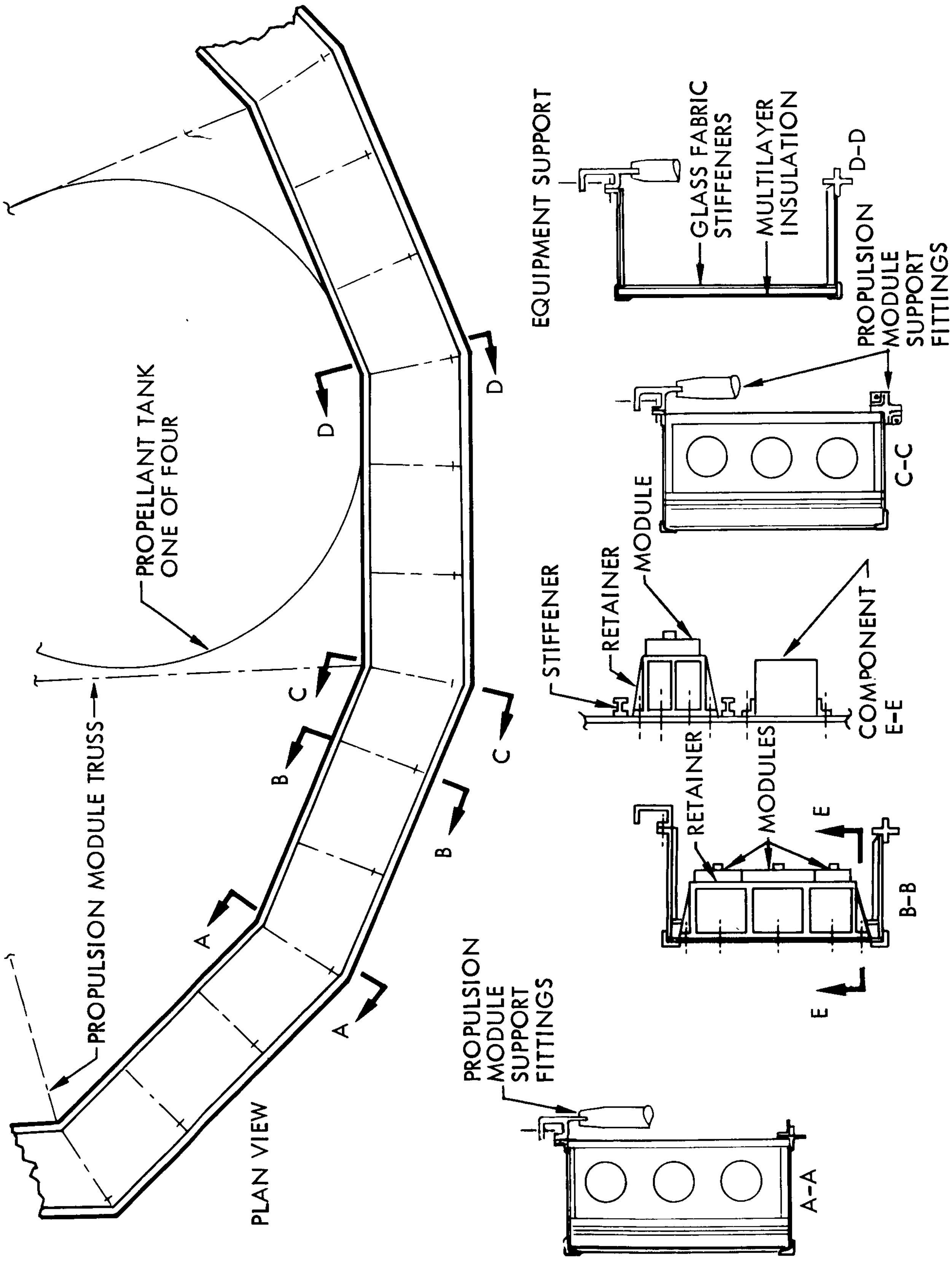


Figure 1.2.10-3: ELECTRONIC MODULES

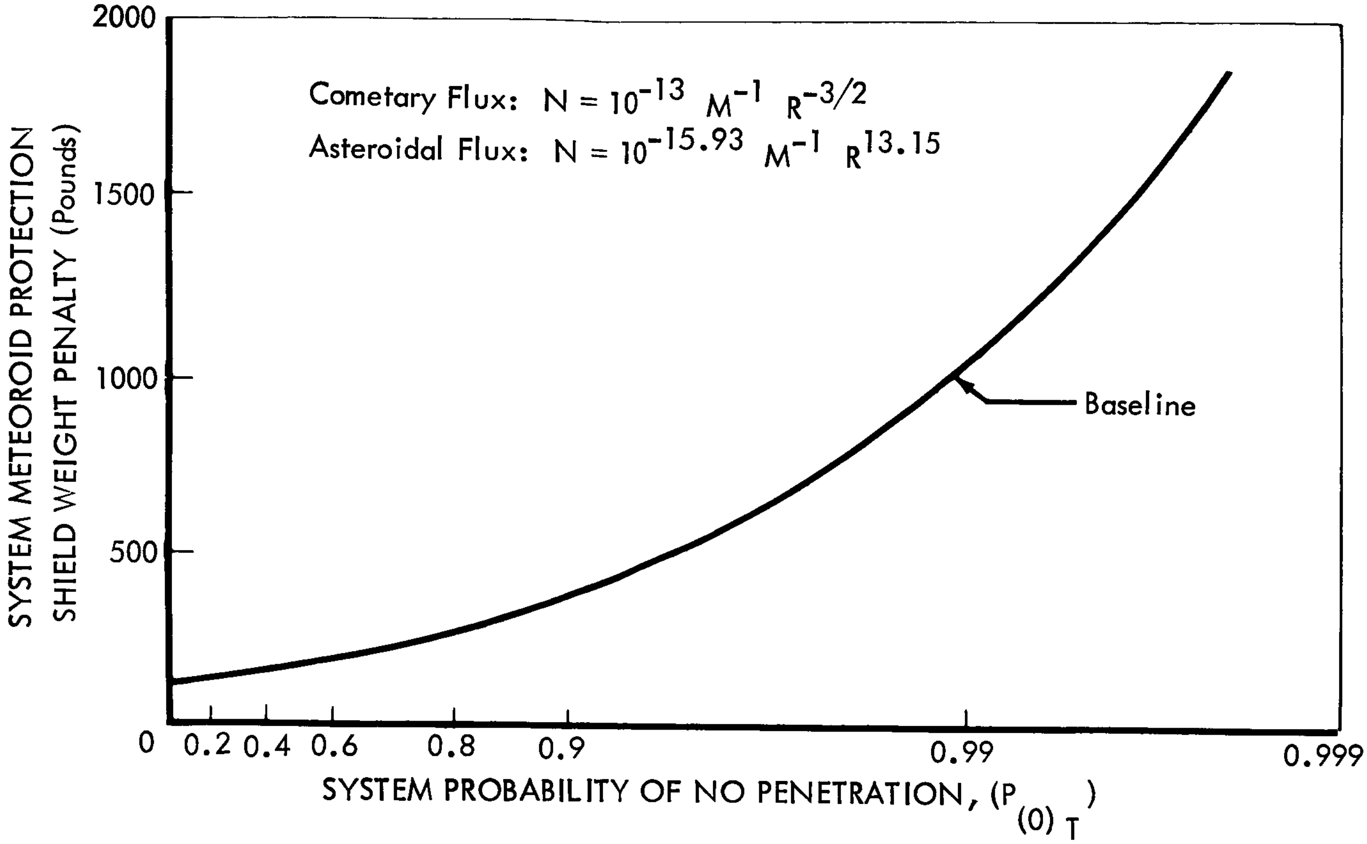


Figure 1.2.10-4: METEOROID SHIELD WEIGHT - 1973 MISSION

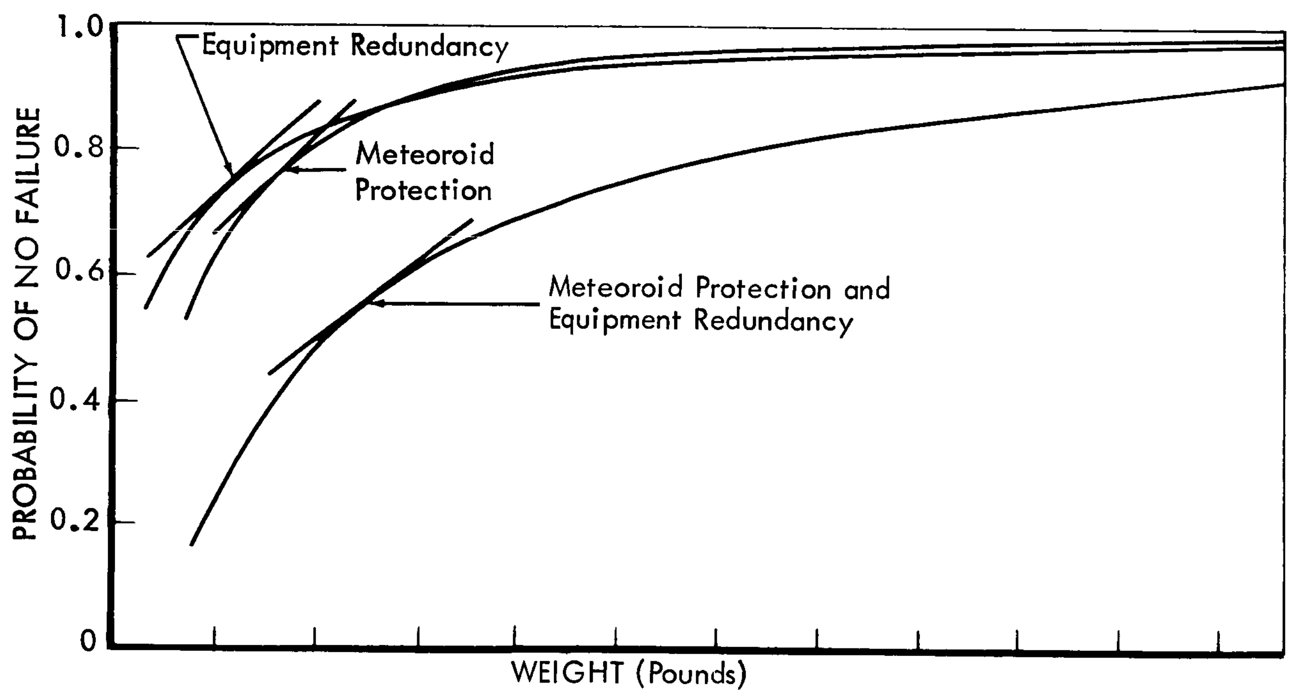


Figure 1.2.10- 5: WEIGHT UTILIZATION TRADES — 1973 Mission

This weight represents an extreme condition assuming a single penetration anywhere within the equipment bay or propulsion tankage bay will cause a failure of the mission. Obviously, the weight penalty to meet this pessimistic criteria is an extravagant use of allocated spacecraft weight. Therefore, the following philosophy which optimizes mission success as a function of meteoroid shielding and/or equipment redundancy is proposed.

The effect of a meteoroid penetration can vary from a catastrophic failure to a slight degradation in the probability of mission success (P_S), depending on the hardware damaged or destroyed. The benefit of weight added for meteoroid protection is evaluated in a manner similar to the method used for adding weight for redundancy. A reference system with no redundancy and no shielding is defined. Weight is then added to redundancy and shielding in those areas and proportions that always yield the greatest possible increase in P_S per pound of added weight. These studies can be conducted jointly in one optimization program, or independently as indicated by the two upper curves on Figure 1.2.10-5. Either approach will yield the resultant lower curve on this figure. If independent studies are conducted, the two resultant curves are combined on a point basis. Pairs of points are selected (one on each curve) where the slopes of the two curves are equal. Each curve represents the maximum probability of no failure that can be achieved for any selected expenditure of weight. The weights corresponding to equal slope points on these curves is the optimum allocation to redundancy and to shielding, and results in the maximum possible P_S for this total weight expenditure. Pairs of points are combined by taking the product of the probabilities of no failure and by adding the two weights. The point on the probability of mission success/weight curve selected for the final design will result from an evaluation of booster performance, spacecraft weight, and spacecraft economics.

<u>Mechanical</u> -- Mechanical system operational functions are shown in Table 1.2.10-3.

Table 1.2.10-3: MECHANICAL OPERATIONAL FUNCTIONS

ITEM	FUNCTIONS: RETAIN (STOWED)	RELEASE	HINGE	DEPLOY	LOCK DOWN	SEPARATION	POSITION CONTROL
High Gain Antenna	X	X	X	X	X		2 Axis
Medium Gain Antenna	X	X	X	X	X		1 Axis
Low Gain Antenna (2)	X	X	X	X	X		
UV Spectrometer Platform							2 Axis
Science Scan Platform	X	X	X	X	X		3 Axis
Planetary Vehicle Separation	X	X				X	

The three major mechanical functions are to retain and deploy, to position-control, and to separate.

1) **<u>Retain and Deploy</u>** -- Mechanical design concepts for spacecraft appendage item retention and deployment are the same as for Task B.

As shown in Figure 1.2.10-6, antennas and the science scan platform are supported in their stowed position, against the loads of launch and boost, by pyrotechnic pinpullers identical to those used on the Lunar Orbiter. Upon command through the pyrotechnic subsystem, the pinpullers are fired, and appendage items are released. Deployment is initiated by the spring compression force stored in the linear actuator. Operating on an appropriate lever arm about the hinge axis, the actuator extends to affect deployment. Appendages are restrained in their deployed position by an integral lock in the actuator. The actuator is identical to ones used on Mariner and Lunar Orbiter programs. Silicone oil is employed to dampen the extension stroke and ensure smooth deployment with minimum dynamic force. Spherical ball bearings, with sleeve bearing inserts and teflon-impregnated fiberglass sliding surfaces, ensure low hinge friction and provide operation redundancy.

2) Position Control -- The high and medium gain antennas, science scan platform, and ultraviolet spectrometer platform are provided with position pointing controls.

   The pointing control mechanism is shown in Figure 1.2.10-7. The pulse drive motor shaft rotates 90 degrees per pulse. Utilizing 900 to 1 speed reduction, the motor steps the boom 0.1 degree per pulse. The position encoder measures 0.044-degree increments of boom rotation. A slew rate of 2 deg/sec is provided. The position controller is housed within an insulated cover in a pressurized, inert gas environment. Internal temperature is controlled by cycling a small electric heater.

   Figure 1.2.10-8 shows the pulse, direction, and feedback electronics for pointing control mechanisms. They are installed in equipment modules with associated electronics.

3) Separation -- The planetary vehicle is attached to the booster shroud by eight explosive bolt/nut assemblies (Figure 1.2.10-9). One assembly is located at each spacecraft-to-shroud structural hardpoint. A separation command from the computing and sequencing subsystem, through the pyrotechnic subsystem, fires an electroexplosive squib on each assembly. Explosive pressure fractures each bolt/nut assembly, releasing the planetary vehicle from the shroud.

   Energy for imparting a minimum relative differential velocity to the planetary vehicle with respect to the booster is stored in eight compression spring assemblies. One spring is located adjacent to each explosive bolt/nut assembly. Each spring is adjusted to provide a minimum tipoff rate to the planetary vehicle during separation.

1.2.10.3 Physical Characteristics

The weight of the structural and mechanical subsystem, shown in Table 1.2.10-4, is based on the configuration shown in Figure 1.1.7-1.

1.2.10.4 Interface Definition

Principal interfaces for the structural and mechanical subsystem are identified in Table 1.2.10-5.

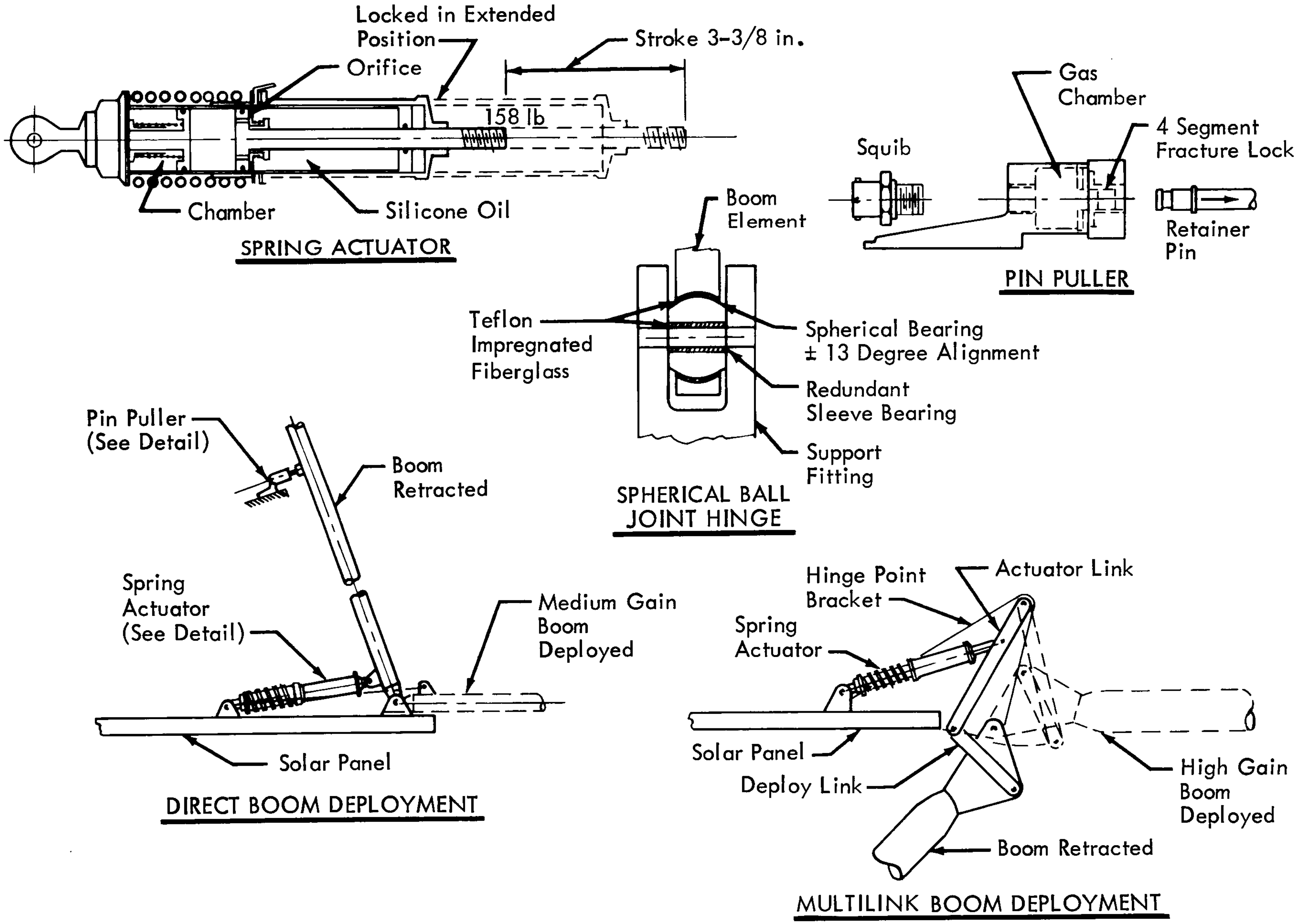


Figure 1.2.10-6: DEPLOYMENT MECHANISMS

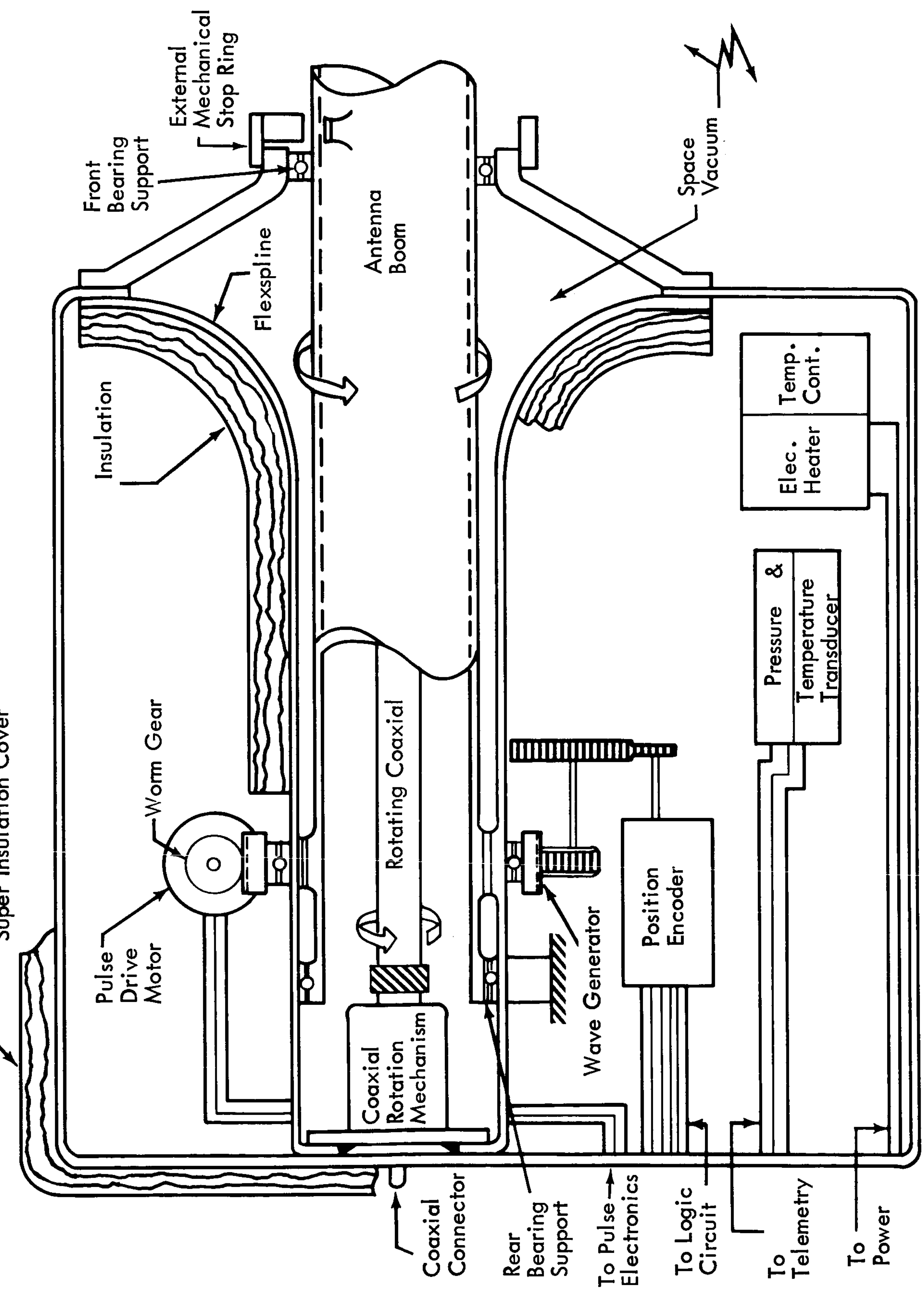


Figure 1.2.10-7: POINTING CONTROL MECHANISM

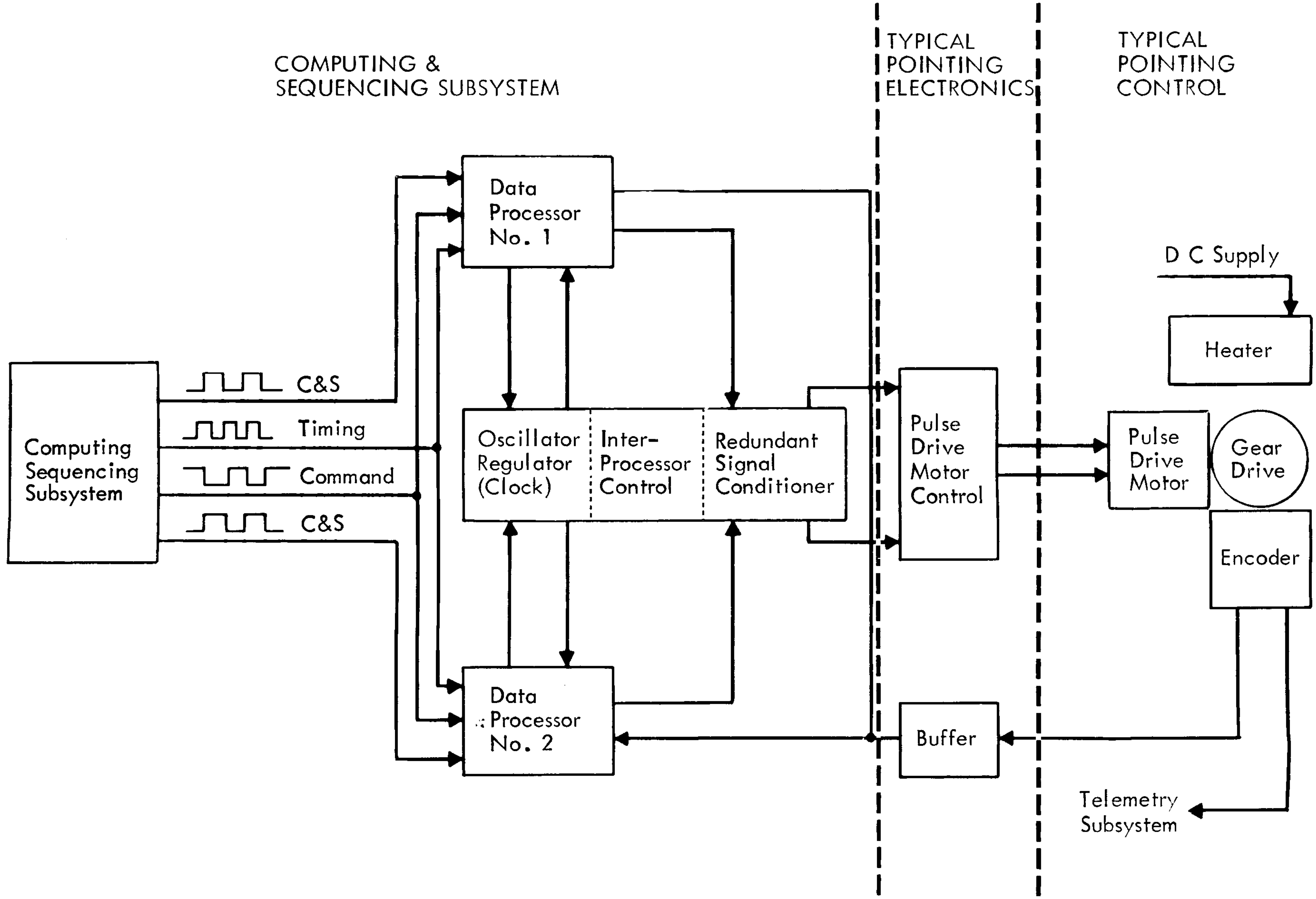


Figure 1.2.10-8: POINTING CONTROL DIAGRAM

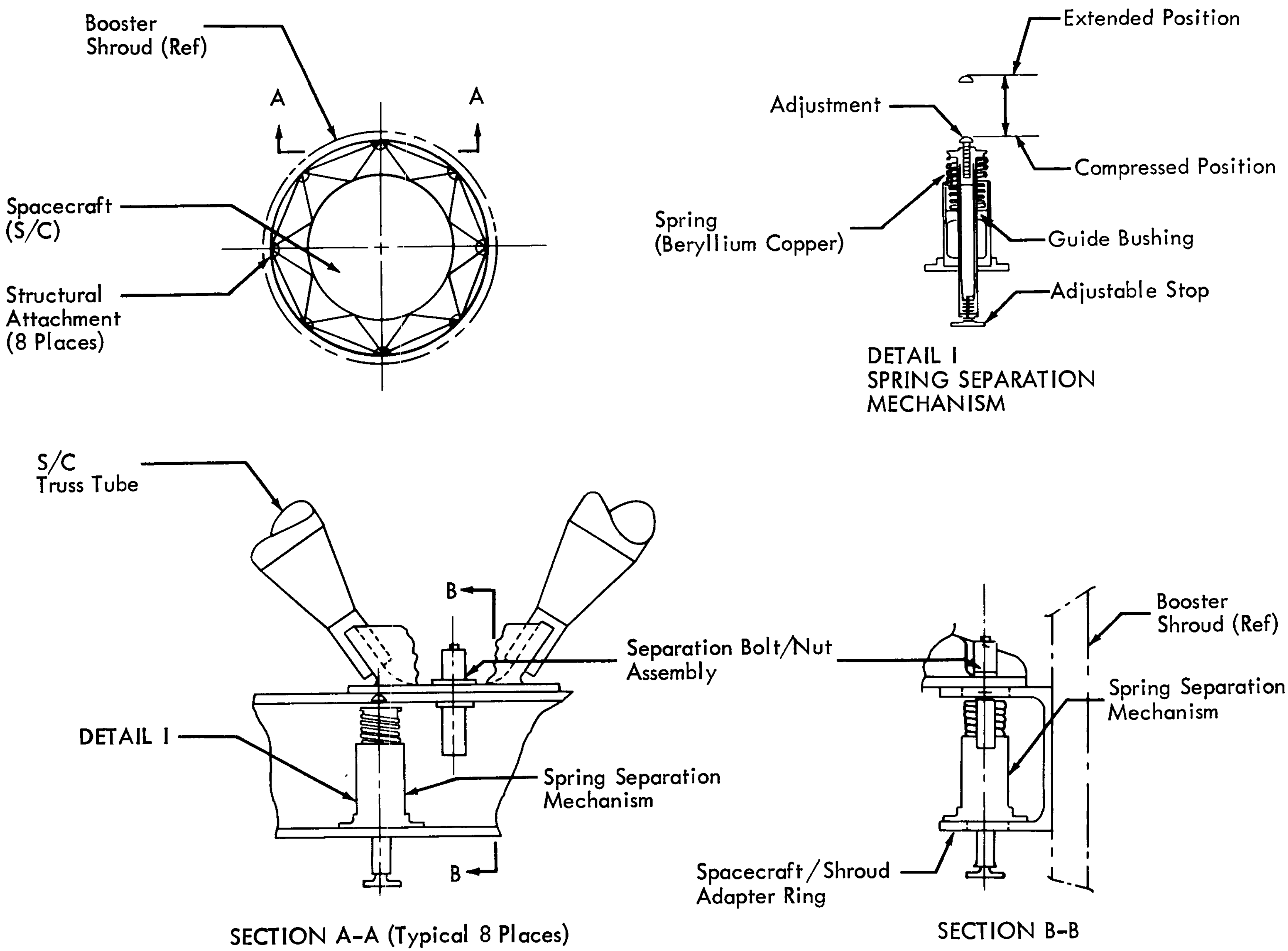


Figure 1.2.10-9: PLANETARY VEHICLE SEPARATION MECHANISM

Table 1.2.10-4: STRUCTURE AND MECHANISM WEIGHTS

	Pounds		
Structure (Basic)			(539)
Capsule Support Truss		129	
Upper Ring/Field Joint Splice	69		
Compression Struts and Fittings (16)	54		
Miscellaneous	6		
Star Truss		318	
Upper Ring	130		
Compression Struts and Fittings (16)	158		
Tension Ties and Fittings	15		
Miscellaneous	15		
Special Purpose Fittings		92	
Capsule Truss/Star Truss	50		
Tiedown/Separation Pads	20		
Miscellaneous	22		
			[1]
Propulsion (Structure)			(1204)
Propellant System		740	
Tanks, Main	543		
Tanks, Bellows	77		
Tank Supports	120		
Pressurization System		440	
Tanks	440		
Tank Supports	(In Propellant System)		
Engine Support and Miscellaneous		24	
LMDE Support	15		
Miscellaneous	9		
Equipment (Structure)			(320.0)
Equipment Compartment		189.0	
Radiator Plates - (24 ft^2)	34.6		
Radiator Plates - (12 ft^2)	43.2		
Equipment Support Stiffeners	10.1		
Secondary Stiffeners	4.8		
Shear Webs	10.6		
Shear Web Chords	8.3		
Cable Tray Webs	26.1		
Cable Tray Outer Chords (2)	11.5		
Cable Tray Inner Chords (2)	15.1		
Miscellaneous	24.7		
Guidance and Control		11.0	
Electronics Assy. Support	11.0		
Instrumentation		76.0	
HGA Boom	25.0		
HGA Support	21.0		
MGA Boom	5.0		
LGA Boom	5.0		
Miscellaneous Supports	20.0		
Electric Power		6.0	
Solar Array Inner Support Fittings	6.0		
Attitude Control System		3.0	
Thruster Supports	3.0		
Science Equipment		35.0	
Scan Platform Supports	32.0		
Cruise Instrument Supports	3.0		
Mechanisms			(120.5)
Antenna Mechanisms		41.5	
HGA - Stowage Mechanism	4.0		
Deployment Mechanism	6.0		
Pointing Mechanism	20.0		
MGA - Stowage and Deployment	3.0		
Pointing Mechanism	4.5		
LGA (2) - Stowage Mechanisms	1.5		
- Deployment Mechanisms	2.5		
Science Mechanisms		30.0	
Scan Platform Pointing Mechanism	20.0		
UV Platform Pointing Mechanism	10.0		
Spacecraft Separation Mechanism		49.0*	

*45 pounds is jettisoned with booster and is included in spacecraft adapter weight provisions.

[1] Weights below are distributed to their respective functional subsystems.

Table 1.2.10-5: STRUCTURAL AND MECHANICAL SUBSYSTEM INTERFACES (Sheet 1 of 3)

ITEM NO.	TYPE OF INTERFACE	INTERFACE DESCRIPTION	INTERFACING SUBSYSTEM	INPUT TO	INPUT FROM	BOUNDARY DEFINITION
1	Structural	Attachment of antenna booms to pointing control	Structure	X		Pointing control adapter attachment holes
2	Structural	Attachment of science scan platform yokes to pointing control	Structure	X		Pointing control adapter attachment holes
3	Structural	Attachment of UV spectrometer pointing controls to structure	Structure	X		Pointing control adapter attachment holes
4	Structural	Deployment hinge bearing installation	Structure	X		Hole and seat in hinge fittings
5	Structural	Attachment of P/V spring thrusters and bolt mechanisms	Structure		X	Controlled surfaces and clearance holes
6	Structural	Attachment of pinpullers	Structure		X	Attachment holes and surfaces
7	Structural	Attachment of pointing control to UV spectrometer	Science		X	Controlled surface and attachment holes
8	Structural	Attachment of pointing control to science scan platform	Science		X	Controlled surface and attachment holes
9	Structural	Low friction bearing on UV spectrometer hinge axis	Structure	X		Bearing receptacle
10	Structural	Low friction bearing on science scan platform hinge axis	Structure	X		Bearing receptacle
11	Structural	Attachment of deployment actuators	Structure		X	Attachment fittings
12	Electrical	Pointing control pulsed power advance and reverse	Computer and Sequencer		X	Connector on pointing control
13	Electrical	Electrical power for pointing control heaters (8)	Electrical Power		X	Connector on pointing control
14	Electrical	P/V separation system energize	Pyrotechnic		X	Electrical connection to EED

Table 1.2.10-5: STRUCTURAL AND MECHANICAL SUBSYSTEM INTERFACES (Sheet 2 of 3)

ITEM NO.	TYPE OF INTERFACE	INTERFACE DESCRIPTION	INTERFACING SUBSYSTEM	INPUT TO	INPUT FROM	BOUNDARY DESCRIPTION
15	Electrical	Energize pinpullers	Pyrotechnic		X	Electrical connection to EED
16	Electrical	Pointing control coaxial cable connection	Telecommunication	X		Connector on pointing control
17	Electrical	Output shaft pointing control position data (8)	Computer and Sequencer	X		Connector on pointing control
18	Temperature	Thermal coatings for actuators and mechanical linkages	Thermal		X	External surface
19	Temperature	Pointing control superinsulation cover	Thermal		X	External surface
20	Structural	Pointing control attachment to high-gain antenna	Telecommunication		X	Bolt pattern
21	Electrical	Structural & Mechanical Subsystem telemetry signals	Telemetry & Booster I.U.	X		Electrical connector at various measurement points
22	Structural	Field joint between capsule and bus. This is a multibolted joint at 160-inch diameter.	Capsule	X		Mounting surface, bolt pattern and size
23	Structural	Field joint between bus and propulsion module. This is an 8-point bolted attachment located approximately on 160-inch diameter.	Bus versus Propulsion	X		Mounting surface, bolt pattern and size
24	Structural	Electronic package attachment to bus.	All flight spacecraft S/S	X		Matched hole pattern between electronic module and bus structure.
25	Structural	Temperature control louver attachment to electronic module frames.	Temperature Control		X	Matched hole pattern between louver assembly and electronic module frame.

Table 1.2.10-5: STRUCTURAL AND MECHANICAL SUBSYSTEM INTERFACES (Sheet 3 of 3)

ITEM NO.	TYPE OF INTERRACE	INTERFACE DESCRIPTION	INTERFACING SUBSYSTEM	INPUT TO	INPUT FROM	BOUNDARY DEFINITION
26	Structural	Science scan platform to bus	Science		X	Mounting surface controlled hole pattern and size
27	Structural	Reaction and control nozzle tanks and tubing support, and attachment to bus	Guidance Control		X	Mounting surface controlled hole pattern and size
28	Structural	Radio antenna mount supports. This includes high-gain antenna, medium gain, (2) low gain and VHF.	Radio		X	Mounting surface controlled hole pattern and size
29	Structural	Guidance scan platform mounting support	Guidance & Control		X	Mounting surface controlled hole pattern and size
30	Structure	Science instruments mounting supports.	Science		X	Mounting surface controlled hole pattern and size
31	Structural	Attitude reference mounting (Canopus, IRU, and fine Sun sensor)	Guidance & Control		X	Mounting surface controlled hole pattern and size
32	Thermal	Control subsystem temperatures throughout all ground test and mission phases. (Refer to Section 1.2.12 for definition of thermal interface design requirements).	Temperature Control		X	Control all subsystem temperatures
33	Structural	Propulsion tanks and engine propulsion support structure	Propulsion	X		Mounting points

1.2.10.5 Reliability

The assessed reliability for the structural and mechanical subsystem and subsystem elements is tabulated in Table 1.2.10-6.

Table 1.2.10-6: RELIABILITY ASSESSMENT
Structures and Mechanisms

SUBSYSTEM ELEMENTS	RELIABILITY ASSESSMENT	RELIABILITY GOAL
Structural Subsystems		
Primary Structure	0.9997	
Support Structure	0.9993	
Mechanical Subsystems		
Planetary Vehicle Separation	0.99978	
Low Gain Antenna Deployment	0.99997	
Medium Gain Ant. Mechanism	0.99545 *	
High Gain Ant. Mechanism	0.99148 *	
Science Scan Platform Mechanism	0.96151 *	
UV Platform Mechanism	0.97467 *	
Total Subsystems Assessment		0.936
Condition 1 - Assume All Elements Are Required.	0.925	
Condition 2 - Medium Gain Ant. Can Back Up High Gain Ant.	0.936	
Condition 3 - Condition 2 plus UV Platform Not Critical	0.960	
Condition 4 - Condition 2 Plus Assume That Either Science Scan Or UV Platform Is Sufficient For Mission Success.	0.997	

* Reliability assessment based on 232 days in transit and 6 months orbital usage.

Subsystem reliability studies have been based on the backup operational mode using the medium gain antenna in the event that the high gain antenna should fail (condition 2 on Table 1.2.10-6). As reflected in condition 4 (Table 1.2.10-6), there is a very high probability that data will be available from instruments on at least one of the two science platforms.

1.2.10.6 Trade Study Summary

Trade studies were performed on the following subjects to support structural and mechanical concept selections:

<u>Structural</u> --
Structural concept analysis (Figure 1.2.10-10).
Structural performance trades (Figure 1.2.10-11).

<u>Mechanical</u> --
Planetary vehicle separation (Figure 1.2.10-12).

The Task B planetary vehicle separation trade study was modified to be consistent with spacecraft configuration changes. Approach (2) "Explosive Nut Release + Spring Separation" is selected as more desirable than (1) "Vee Block Band +Spring Separation," primarily because its shroud interface is much less complicated.

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT

TRADE STUDY NUMBER & TITLE STRUCTURAL CONCEPT ANALYSIS

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

The bus primary structure must provide support for electronic packages, transfer flight capsule loads to an adapter, interface with propulsion support structure and perform other secondary structural functions.

Design loads occur during boost. The purpose of this study is to compare 6 basic structural configurations, involving both separate and integral adapters, to determine the most efficient structural configuration.

Trade study ground rules are:

(1) Semimonocoque concepts will use non-compression buckling, shear-resistant skin.

(2) All aluminum structure.

(3) Concentrated loads at 8 points using doublers for load distribution.

(4) Combined load condition using limit load factors of 4.75 axial and 2.0 lateral.

(5) An ultimate factor of safety of 1.25.

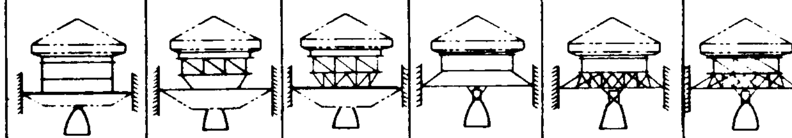
(6) A flight capsule weight of 5000 pounds.

(7) All columns shall consider 1% eccentricity to account for manufacturing and thermal misalignment.

(8) Bus diameter is 160 inches.

MATRIX OF DESIGN APPROACH

STRUCTURAL CONCEPT



	1	2	3	4	5	6
	SEPARATE ADAPTER SEMIMONO	SEPARATE ADAPTER SEMIMONO & TRUSS	SEPARATE ADAPTER ALL TRUSS	INTEGRAL ADAPTER SEMIMONO	INTEGRAL ADAPTER SEMIMONO & TRUSS	INTEGRAL ADAPTER ALL TRUSS
BUS PRIMARY STRUCTURE	459 lb	336 lb	224 lb	854 lb	520 lb	408 lb °
PROP. PRIMARY STRUCTURE	114 lb	201 lb	201 lb	114 lb	114 lb	114 lb
SECONDARY STRUCTURE	412 lb	453 lb	473 lb	419 lb	411 lb	431 lb
TOTAL STRUCTURE IN MARS ORBIT	985 lb	990 lb	898 lb	1387 lb	1045 lb	953 lb

SELECTION

Results

- Concepts 1 & 4 are the heaviest primary structure.
- Concepts 3 & 6 are the lightest primary structure.
- Trusses give the lightest total structure in Mars orbit for both integral and separate adapter configurations.

SELECTED APPROACH

Based on minimum structural weight in Mars orbit, trusses were selected for the baseline configuration.

Figure 1.2.10-10: STRUCTURAL CONCEPT TRADE STUDY

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE: STRUCTURAL PERFORMANCE TRADES	SELECTION
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS		MATRIX OF DESIGN APPROACH	

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

The bus primary structure must provide support for electronic packages, transfer flight capsule loads to an adapter, interface with the propulsion support structure and perform other secondary structural functions.

Provide minimum weight to meet performance requirements.

Trade Study Ground Rules:

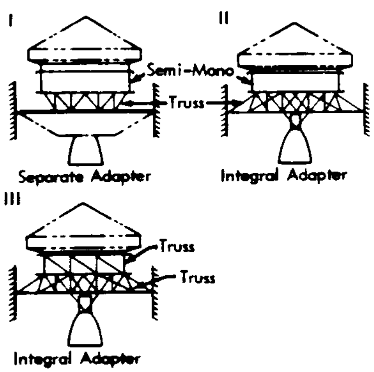
(1) Bus diameter is 160 inches.

(2) Colums are sized for 1% eccentricity to account for thermal and mfg. misalignment.

(3) Semimonocoque structure is shear resistant and noncompression buckling.

(4) Ultimate factor of safety of 1.25.

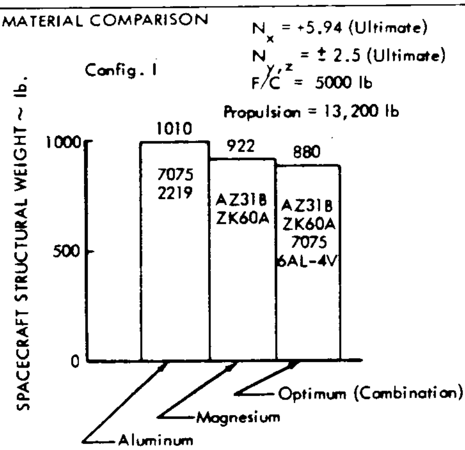
(5) The following configurations are traded.



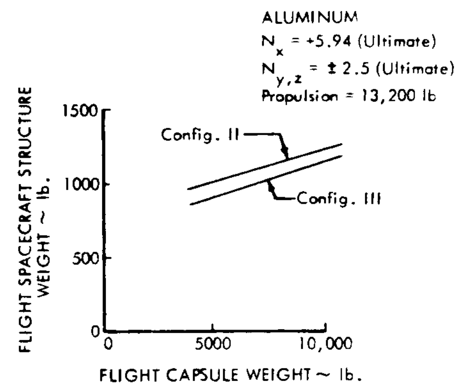
MATRIX OF DESIGN APPROACH

To select a design approach and fully evaluate the effects of material selection, changes in flight capsule weight, and effect of loads, the following trades are performed:

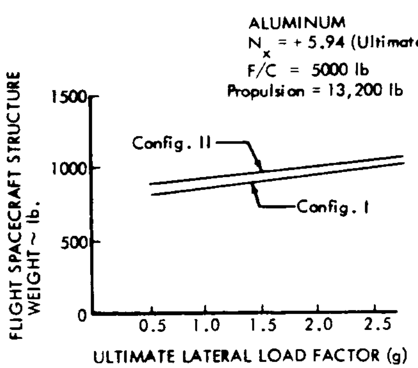
(1) MATERIAL COMPARISON



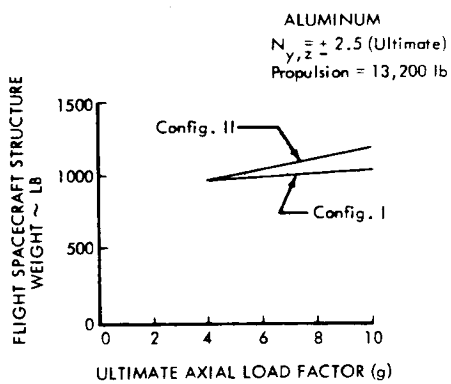
(2) EFFECT OF FLIGHT CAPSULE WEIGHT



(3) EFFECT OF LATERAL LOAD FACTOR



(4) EFFECT OF AXIAL LOAD FACTOR



SELECTION

Material

Combined material's are lightest.

- Aluminum is heaviest.
- Total weights are close.
- Magnesium is most expensive and complex to work.

F/C Weight Effect

- Small effect for range of design interest. ~ 10%

Load Factor

- Small effect for range of design interest

SELECTED APPROACH

Material

Aluminum based on cost & mfg simplicity.

Hold combined load factors as design condition.

Figure 1.2.10-11: STRUCTURAL PERFORMANCE TRADE STUDY

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE	PLANETARY VEHICLE (P/V) BOOSTER SEPARATION

MATRIX OF DESIGN APPROACH

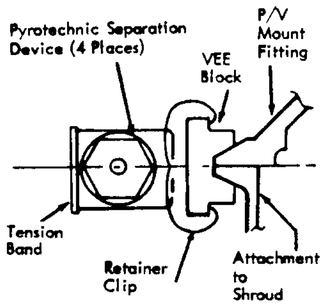
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

1. Simple, conservative design and space-proven components to be used whenever possible.
2. Design to provide minimum uncertainty of predicting actual separation velocity and tipoff rate.
3. The probability of successful separation will be at least 0.999+ to ensure mission success.
4. EED's will be single cartridge dual bridge wire devices and products of combustion shall be wholly contained.
5. Personnel hazard factor will be 1.76.

The competing characteristics traded are:

Reliability
Integral weight
Growth potential
Installation simplicity
Space-proven concept
Tipoff rate
Shroud interface

1. VEE BLOCK BAND RELEASE SPRING SEPARATION



A four-segment tension band system is coupled by turnbuckle fittings. Pyrotechnic devices, one in each band segment, fire simultaneously for uniform band release. Energy stored in 8 separation thruster springs provides P/V separation forces.

DISCUSSION

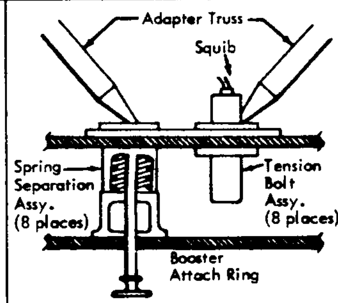
PRO

1. Concept (with smaller diameter tension bands) space-proven on L.O. & Mariner programs.
2. Reliability 0.99992.
3. Provides reasonable control of separation velocity & tipoff rate

CON

1. Band installation requires strain gages and associated instrumentation.
2. Requires protective shield for personnel during installation.
3. P/V installation is complex.
4. Large band assembly must be retained after release.
5. Integral weight of release device 1 pound on P/V and 56 pounds on booster.
6. Critical interface with booster shroud.

2. EXPLOSIVE NUT RELEASE + SPRING SEPARATION



Eight explosive nut release assemblies fasten the P/V to the booster shroud. EED activation, through the pyrotechnic subsystem, fractures a tension fastener. Tapered pins provide alignment and shear capability. Energy stored in 8 separation thruster springs provide P/V separation forces.

DISCUSSION

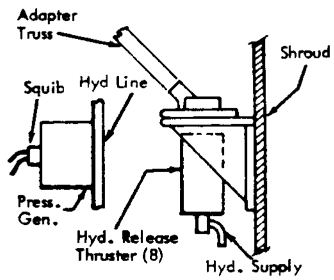
PRO

1. Concept space-proven on Burner II & other Space/Flight programs.
2. Strain gages, and associated instrumentation not required.
3. Integral weight of release device 3 pounds on P/V & 46 pounds on booster.
4. Reliability 0.99968
5. Noncritical shroud interface.
6. Provides reasonable control of separation velocity & tipoff rate.

CON

1. Requires simultaneous firing & separation of all 8 electro-explosive devices.
2. Moderately complex P/V install.

3. HYDRAULIC RELEASE & SEPARATION



Eight thrust fittings are activated by a hydraulic manifold system. Fluid, pressurized by a pyrotechnic gas generator, unlocks each thruster and forces its extension to provide P/V separation and tipoff rate control.

DISCUSSION

PRO

1. Simple, conservative design concept.
2. Permits off-center P/V c.g. location without causing excessive tipoff rate.
3. Provides good control of separation velocity.
4. P/V installation is simple.
5. Strain gages associated instrumentation not required.
6. Permits simplified release tests to be made after P/V installation.
7. Reliability 0.99994.

CON

1. System not previously spaceflight proven.
2. Integral weight of release device 4 pounds on P/V and 57 pounds on booster.
3. Shroud interface includes pyrotechnic cable.

SELECTION

1. Reliability 3, 1, 2
2. Integral Wt. on P/V 1, 2, 3.
3. Growth potential 3, 2, 1.
4. P/V installation simplicity 3, 2, 1.
5. Concept space proven on L.O. Mariner, Burner II, & other space-flight programs 1, 2.
6. Tipoff rate 3, 2, 1.
7. Shroud interface 2, 3, 1

SELECTED APPROACH

(2) Explosive nut release & spring separation.

Figure 1.2.10-12: PLANETARY VEHICLE/BOOSTER SEPARATION TRADE STUDY

A new approach (3), "Hydraulic Release and Separation," has many advantages. It was not selected because it is not spaceflight proven. However, it is a simple, conservative design based on standard technology and analysis of this system will continue.

1.2.10.7 New Technology and Development Items

No new technology or development is anticipated for the structural and mechanical subsystem.

1.2.10.8 Growth Potential

Structural -- The flight spacecraft is structurally sized to carry maximum loads occurring on 1973 through 1979 missions. The flight spacecraft can therefore carry up to a 7000-pound flight capsule. The propulsion tank volume and tank support structure are adequate to carry the maximum fuel volume and loads occurring on 1973 through 1979 missions. Reduction in flight capsule weight may be placed in electronics or science without modification to primary structure.

Mechanical -- Spacecraft mechanisms are designed specifically for the 1973 mission. If subsequent mission requirements can be identified with certainty during the 1973 mission design phase, the desirability of incorporation will be evaluated.

1.2.11 Pyrotechnic Subsystem

1.2.11.1 Design Constraints and Requirements
1.2.11.2 Functional Description and Performance Characteristics
1.2.11.3 Physical Characteristics
1.2.11.4 Interface Definition
1.2.11.5 Reliability and Safety
1.2.11.6 Trade Study Summary
1.2.11.7 New Technology and Development Items
1.2.11.8 Growth Potential

1.2.11 Pyrotechnic Subsystem

1.2.11.1 Design Constraints and Requirements

The pyrotechnic subsystem complies with AFETRM 127-1, General Range Safety Plan. Constraints and requirements are summarized below.

1) Redundancy is provided for critical functions.

2) Pyrotechnic devices are nonventing and nonrupturing, and provide positive gas containment.

3) No failure mode (including procedural deviation) will both arm and command the pyrotechnic subsystem.

4) Conversion of a.c. to d.c. is done in the pyrotechnic power switching unit.

5) EED firing circuits are designed to provide continuous circumferential shielding and to ensure that when connectors are mated the shield connection is made before firing circuit pins are contacted.

6) Pyrotechnic circuit ground point is located within the pyrotechnic power switching unit and connects to its chassis at one point only.

7) Redundant firing circuits are provided.

1.2.11.2 Functional Description and Performance Characteristics

The functional schematic diagram is shown in Figure 1.2.11-1. Pyrotechnic functions are controlled by the computer and sequencer.

Pyrotechnic Power Switching Unit (PPS) -- This unit contains power supplies, bridgewire drivers, interface circuits, and power "on-off" latching relays. It contains two separate transformer-rectifier (T-R) units; two sets (two each) of capacitor banks, and two independent paths (each consisting of a driver-bridgewire circuit). Each T-R is associated with a set of firing circuits designated circuit A and circuit B. Energy storing capacitors are used to allow simultaneous firing of several functions. Capacitor banks are time-shared to reduce weight, volume, and power drain. Solid state switches are used in EED drivers and associated amplifiers, and in each EED bridgewire circuit logic gate.

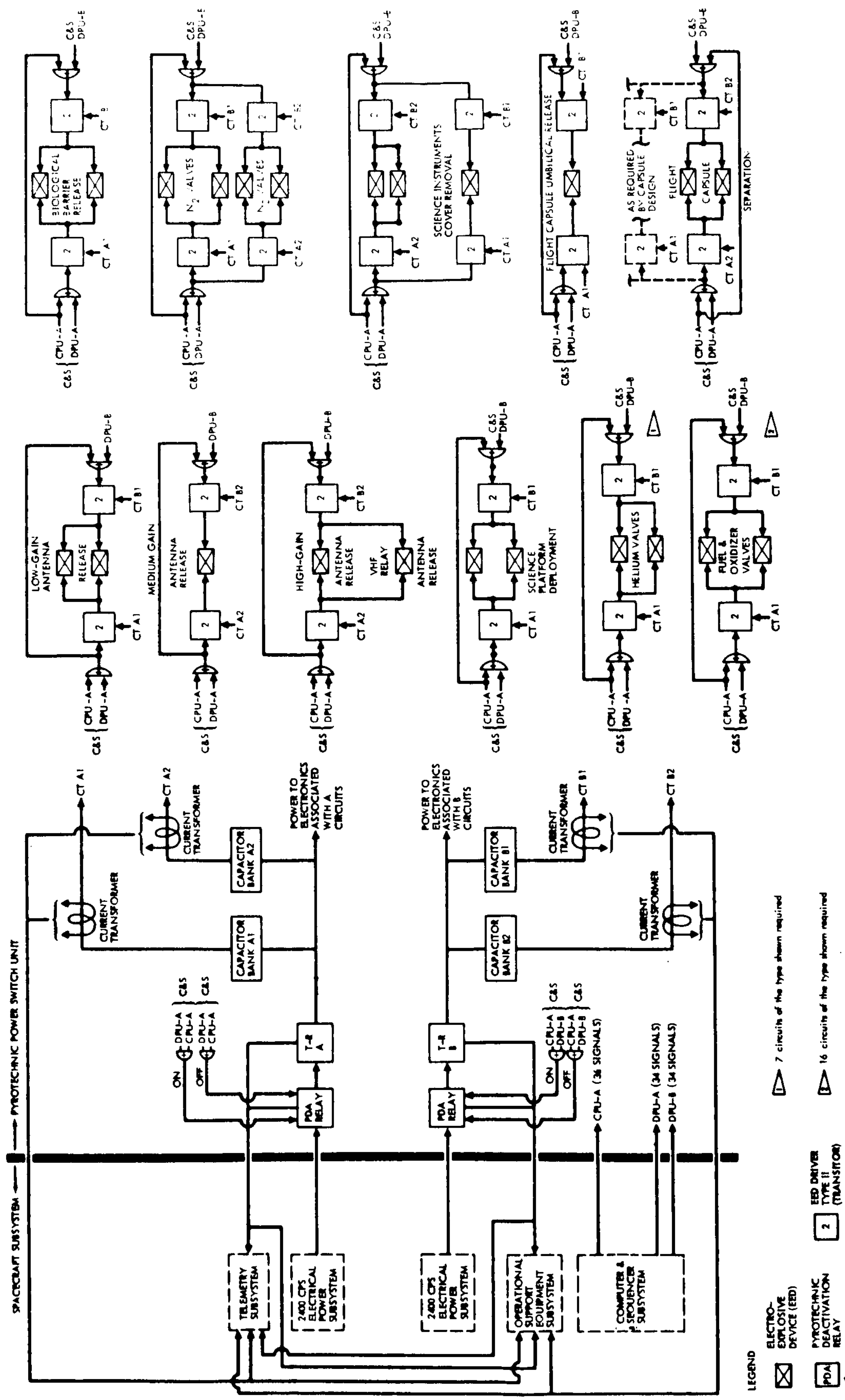


Figure 1.2.11-1: PYROTECHNIC SUBSYSTEM SCHEMATIC DIAGRAM

Latching relays provide power "turn on" and "turn off" capability. Prelaunch safe/arm signals to the OSE and to telemetry evaluate the pyrotechnic subsystem operating condition.

The pyrotechnic driver circuit (Figure 1.2.11-2) was chosen as a result of trade studies. It ensures that no single failure will cause premature, delayed, or inhibited functioning. A single failure in either leg may turn the output circuit in one leg on. However, the EED will receive no activating current because the other output circuit is off. The redundant drivers are used for all functions to avoid a single off failure.

Capacitor Bank Circuit (CT) -- This circuit (Figure 1.2.11-1) consists of parallel capacitors charged through a series resistor. A diode prevents capacitor discharge from transistor leakage. Resistors provide leakage current to parallel circuits. The computing and sequencing subsystem provides input signals of 4.8 kHz, 100-millisecond duration, and 6-volt amplitude. The power subsystem provides 2400 cps, 50-volt amplitude power. The outputs are electrical energy pulses to fire EED's, signals to the OSE, and data to the telemetry subsystem.

1.2.11.3 Physical Characteristics

Pyrotechnic subsystem weight is shown below.

Item	Quantity Required	Total Weight (pounds)
Pyrotechnic power switching unit	1	50
EED's	71	5
Structural frame	1	4
Total		59

1.2.11.4 Interface Definition

Principal interfaces for the pyrotechnic subsystem are identified in Tables 1.2.11-1 and 1.2.11-2.

1.2.11.5 Reliability and Safety

Reliability -- The pyrotechnic subsystem utilizes redundant paths to achieve a high reliability. At least two failures must occur to prevent EED firing, or to cause inadvertent EED firing in any function. Reliability is assessed at 0.9999.

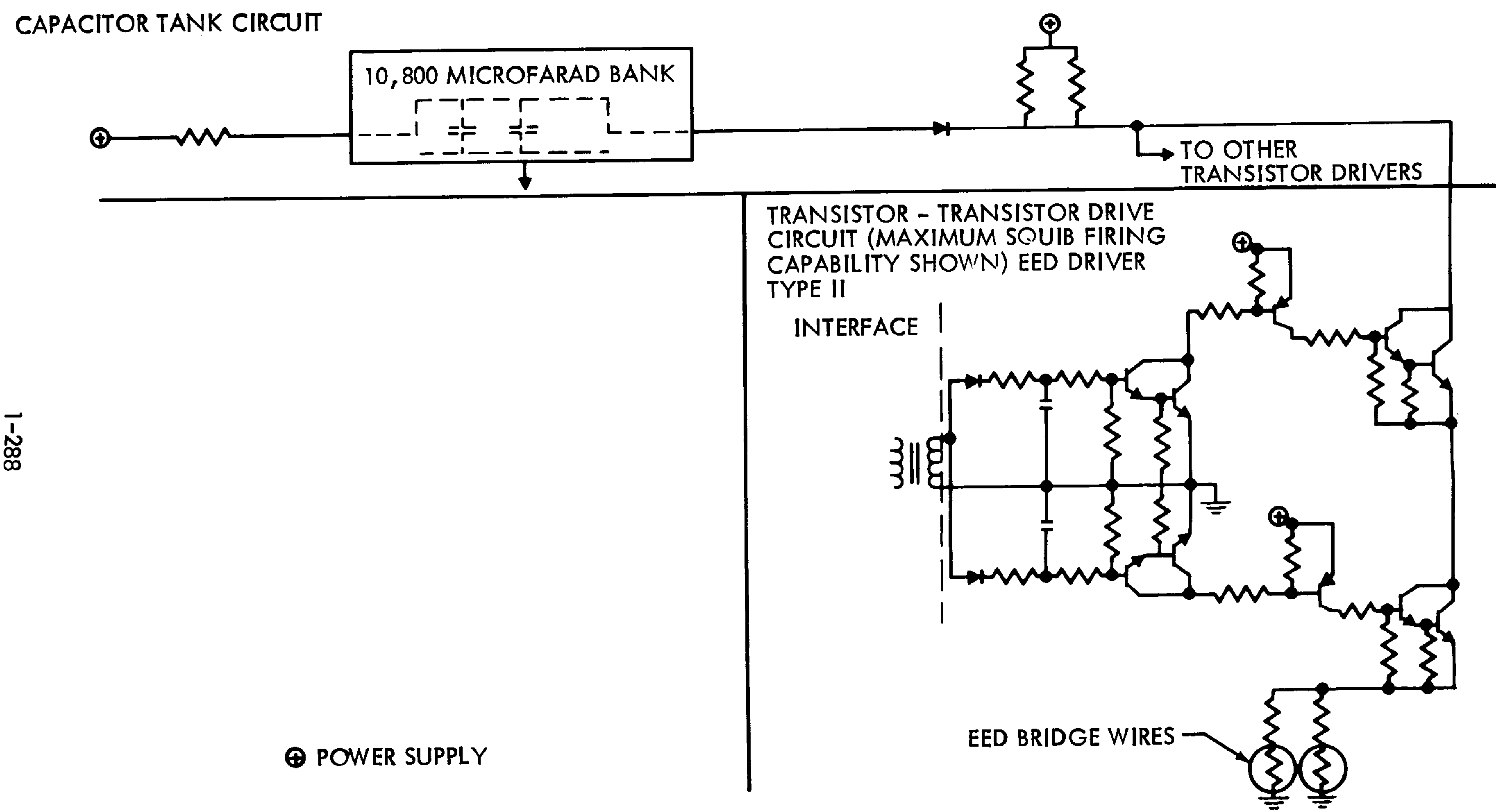


Figure 1.2.11-2: PYROTECHNIC DRIVER CIRCUITS

Table 1.2.11-1: PYROTECHNIC SUBSYSTEM INTERFACES

ITEM NO.	TYPE OF INTERFACE	INTERFACE DESCRIPTION	INTERFACING SUBSYSTEM	INPUT TO	INPUT FROM	BOUNDARY DEFINITION
1	Structural	Installation of power switching unit	Structural & mechanical	X		Mounting points
2	Thermal	Control subsystem temperatures throughout all ground test and mission phases.	Temperature control		X	Control all subsystem temperatures
3	Electrical	Pyrotechnic subsystem telemetry signals (10 measurement)	Telemetry	X		Cable connector at pyrotechnic subsystem
4	Electrical	Power input, 50 volts rms, KHZ, bus B & C	Power		X	Cable connector at pyrotechnic subsystem
5	Electrical	Pyrotechnic subsystem command signals (36 commands)	Command		X	Cable connector at pyrotechnic subsystem
6	Electrical	Input commands signals for initiating EED's (68 commands)	Computing & sequencing		X	Cable connector at pyrotechnic subsystem
7	Electrical	Planetary Vehicle separation signal	Computing & sequencing	X		Cable connector at pyrotechnic subsystem
8	Electrical	Between the pyrotechnic subsystem and other subsystems shown on this table and Table 1.2.11-2.	Noted of Figure 1.2.11-1 & Table 1.2.11-2	X	X	Cable connector at pyrotechnic subsystem

Table 1.2.11-2: PYROTECHNIC FUNCTIONS AND EED INTERFACES

ITEM	EQUIPMENT	INTERFACING SUBSYSTEM	NO. OF EED'S
1	Low-gain antenna release (2 booms)	Structural & mechanical	2
2	Medium-gain antenna release	Structural & mechanical	1
3	High-gain antenna release	Structural & mechanical	1
4	VHF relay antenna release	Structural & mechanical	1
5	N_2 valves	Reaction control	4
6	Science scan platform release	Structural & mechanical	1
7	Science instrument covers removal	Science	3
8	Helium valves	Propulsion	14
9	Fuel valves	Propulsion	16
10	Oxidizer valves	Propulsion	16
11	Biological barrier release	Capsule	2
12	Flight capsule umbilical release	Capsule	1
13	Flight capsule separation	Structural & mechanical	8

Safety -- Safety is enhanced by the conservative approach taken to handle subsystem status monitoring, arm/disarm functions, RF signals, and transient voltages. Two latching relays, one in series with each power supply circuit to the pyrotechnic subsystem, perform the the arm/disarm function. These relays can be commanded "on" either just prior to launch or after spacecraft separation. Status monitoring signals are sent to the OSE during prelaunch and to telemetry during flight. Capacitor banks connected to the drive circuits act as filters to stray RF signals and any induced transient voltages.

1.2.11.6 Trade Study Summary

Pyrotechnic Power Switch -- A trade study (Figure 1.2.11-3) was performed to support the selection of a safe/arm concept for the pyrotechnic subsystem. The design approach "computing and sequencing subsystem by command" is the selected approach.

1.2.11.7 New Technology and Development Items

No new technology or development is anticipated for the pyrotechnic subsystem.

1.2.11.8 Growth Potential

The pyrotechnic subsystem can easily be modified to support anticipated requirements for subsequent missions.

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE: PYROTECHNIC SAFE/ARM DEVICE — MATRIX OF DESIGN APPROACH		SELECTION
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS		Pyrotechnic Arming Switch (PAS) and Separation Initiation Timer (SIT)	Computing and Sequencing Subsystem by Command	
1) Power to the pyrotechnic subsystem shall be blocked by the normally open contacts of a safe/arm device. 2) No single or common failure mode (including procedural deviation) shall both arm and command the pyrotechnic subsystem. Competing characteristics traded are: a) Reliability - performance b) Power c) Weight/volume d) Cost e) Design complexity		Approach No. 1 At planetary vehicle separation this approach provides: a) A mechanical device that closes circuits to arm the pyrotechnic subsystem by connecting to spacecraft power. b) OSE prelaunch "safe/arm" indication c) "Arm" signal to the telemetry subsystem d) Switching sequence to arm the pyrotechnic subsystem, e) A signal to the C&S that the planetary vehicle has separated. Discussion PRO: 1. Number of failures during the mission = 49.7×10^{-6}. 2. No power required except for brief period.. 3. SIT is a simple mechanized/electrical timer. CON: 1. Weight = 1 pound, volume = 108 cubic inches 2. Cost = $1,000 to $5,000. 3. Does not allow testing of PAS and SIT on the launch pad. 4. Requires electromechanical interface with the booster.	Approach No. 2 In view of the redundancy in the C&S, its ability to provide timed back-up functions, and its separate command path back-up; it is unnecessary to provide an internal separate set of hardware to provide safe/arming of the Pyrotechnic subsystem. Both real time commands from the ground and stored program commands can initiate arming of the Pyrotechnic subsystem after separation. Discussion PRO: 1) Performance equivalent to #1. Effect on reliability due to addition of the following parts is negligible. a) Decoding gates and interface amplifiers. b) 2 micro circuit flatpacks. c) 4 amplifiers. (Anticipated failure rate = 0.82×10^{-6}) 2) Weight = 1/8 pound, volume = 3 cubic inches. 3) Cost = $100 - $500. 4) No electromechanical interface with the booster. 5) Increased pyrotechnic subsystem checkout flexibility on launch pad. CON: 1) Power = 0.08 watts continuous due to increase in function decoding. 2) Already developed circuitry concept is simpler. More complex operationally.	Reliability-Performance 2, 1 Power 1, 2 Weight/Volume 2, 1 Cost 2, 1 Design Complexity 1, 2 SELECTED APPROACH No. 2

Figure 1.2.11-3: PYROTECHNIC SAFE/ARM DEVICE TRADE STUDY

BLANK

1.2.12 Temperature Control Subsystem

1.2.12 Temperature Control Subsystem

The temperature control subsystem consists of bimetal actuated louvers, multilayer thermal insulation systems, and surface coatings and finishes. The spacecraft mass is used as a heat sink during transients. The Voyager temperature control concepts and components are based upon Mariner IV and Lunar Orbiter design and experience. The primary means of temperature control is via the radiator-louver assemblies. Functional redundancy is achieved by separately actuated louver blades.

1.2.12.1 Design Constraints and Requirements

The design constraints and requirements for the temperature control subsystem are presented below:

1) Variation of the solar intensity (136 to 47 watts/ft^2).

2) Nose fairing heating during ascent and Earth orbit park, (65 to 40 watts/ft^2).

3) Solar occultation at Mars up to 1.5 hours.

4) Maximum time between launch and Sun acquisition is 150 minutes.

5) Planetary vehicle maneuvering with respect to the solar vector for up to 60 minutes.

6) Equipment power profile (See Section 1.2.1).

7) Passive thermal control.

8) Design Conditions - Table 1.2.12-1 presents a list of all temperature control components and the characteristics that dictate their design.

9) Temperature Limits - Normal equipment operating limits are based on results of existing reliability test data and are selected in the region of low equipment failure rates and long battery life. Figure 1.2.12-1 indicates minimum failure rates occur at about 289 to 292°K (60 to 65°F). Radiator temperature control range is 280°K (45°F) to 297°K (75°F); 17°K (30°F) is a reasonable control range for bimetal actuated louvers. Figure 1.2.12-2 indicates the expected battery life as a function of temperature. The battery bay thermal control range is 269°K (25°F) to 286°K (55°F).

10) Environmental Control on Launch Pad - After encapsulation in the nose shroud, gas will be ducted into nose fairing cavity for providing positive pressure, temperature, and humidity conditioning. During prelaunch operations, gas at bulk temperature between 280°K (45°F) and 292°K (65°F) will be provided through the shroud umbilical. Just prior to liftoff, spacecraft bus equipment will be precooled to 280±3°K (45±5°F). Bus cooling requirements on the pad will not require use of cooling gases at differing temperatures ducted preferentially to specific subsystems.

Table 1.2.12-1: TEMPERATURE CONTROL DESIGN CONDITIONS

		REQUIREMENTS								
		Ground Phases		Flight Phases						
Component	Design Characteristic	Testing ICDM TAT FAT	Prelaunch	Launch to Interplanetary Cruise	Cruise Near Earth	Midcourse Near Earth	Late Cruise	Insertion Into Mars Orbit	Mars Orbit	Mars Orbit Solar Occultation
Louver-Radiator Assembly	Area				Max Area					Min Area
Thermal Mass (Inertia)	Mass, Specific Heat			Total Bus Equip.		Individual Components				
Radiator Plate	Conductance								Temp. Gradients	
Radiator Plate - TWTA - Battery	Conductance & Diffusivity								Max Heat Load	
Coating - Radiators	Absorptance, Emittance				Max Emittance	Min Absorptance				
Coating - Internal Equip. & Structure	Emittance			Mass Differences					Temp Gradients	
Thermal Shield - Side Panels	Conductance			Min Heat In						Min Heat Out
Insulation - Bus - Propulsion Module Isolation	Conductance				Min Heat In				Min Heat Out	
Coating - Side Panel	Temperature	Max Predictability								Min Heat Out
Isolation Joint - Booster & Solar Panel Attachments	Conductance				Min Heat In					
Isolation Joint - Propulsion Module	Conductance				Min Heat In				Min Heat Out	
Isolation Joint - Solar Panel Struts	Conductance									Min Heat Out
Isolation Joint - Capsule Attachment	Conductance									Min Heat Out
Ground Cooling Duct	Gas Flow & Temperature		45 $\pm$ 5°F							
Thermal Shield - Guidance & Control Bay	Conductance				Min Heat In					Min Heat Out
Thermal Shield - Gimbal Actuators	Conductance				Min Heat In					Min Heat Out
Coating - Booms & Antennas	Temperature	Max Predictability								
Engine Heat Shield	Area, Emittance							Minimum Solar Array Heating		

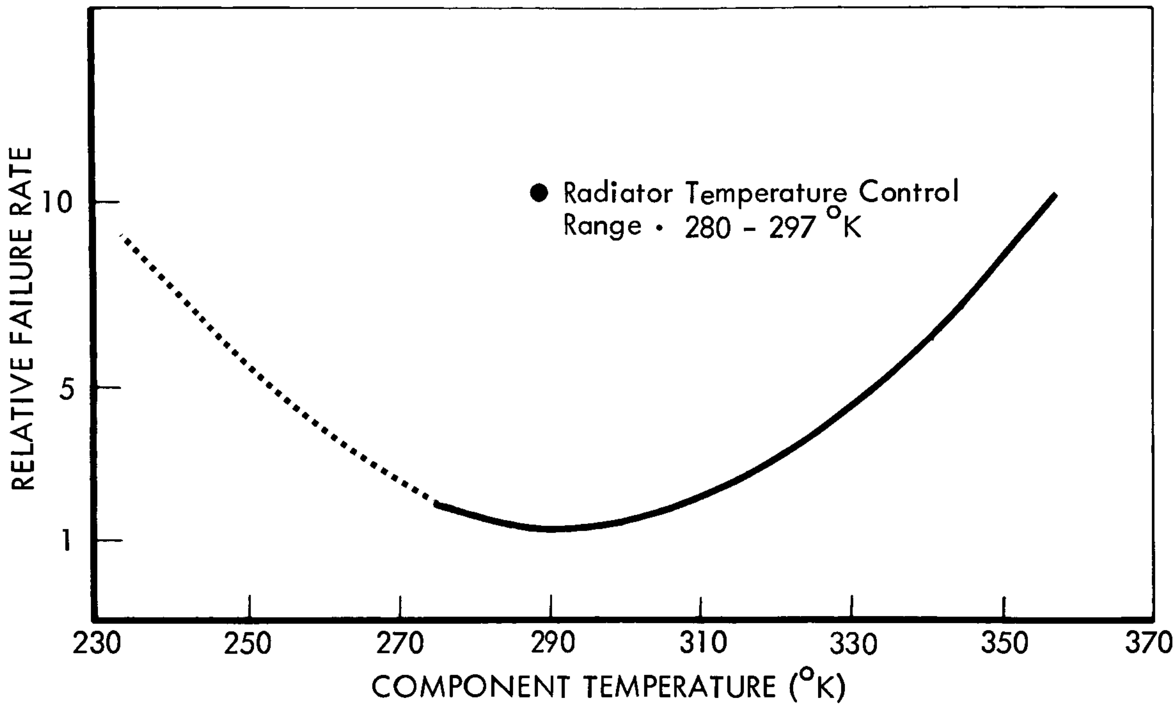


Figure 1.2.12-1: ELECTRONIC EQUIPMENT FAILURE RATES

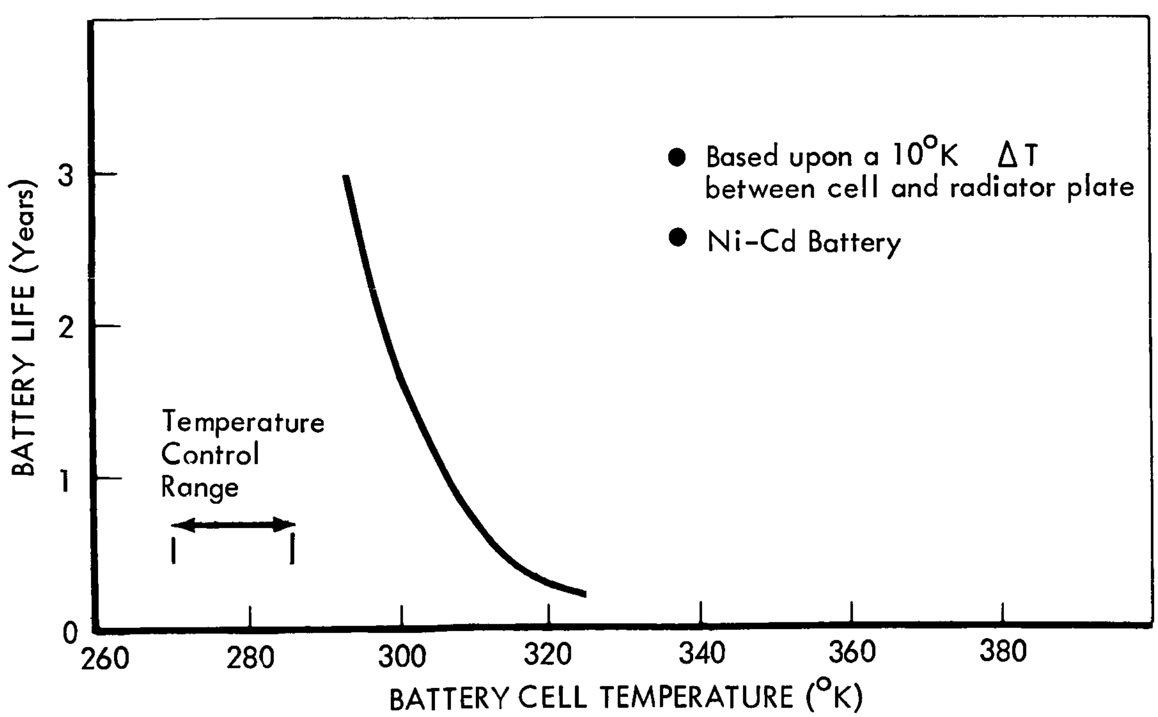


Figure 1.2.12-2: BATTERY CELL LIFE

11) Solar array maximum temperature is 423°K (300°F), minimum temperature is 99°K (-282°F), and maximum temperature gradient is 2.25°K/sec (4°F/sec).

12) Ascent Venting - The temperature control subsystem will withstand an instantaneous pressure decrease of 0.0277 inches H_2O during nose fairing separation (controlled nose fairing venting during ascent will be provided by the launch vehicle). All bus insulation panels and thermal radiation shields will be adequately vented to prevent damage during ascent.

13) Propulsion module temperature limits are established by the physical properties of the propellants and are 278°K (40°F) to 317°K (110°F).

14) The maximum heat flux from the LMDE nozzle extension is equivalent to a black body at 867°K (1100°F) or 2971 watts/ft^2.

15) Reliability allocation is 0.9999.

1.2.12.2 Functional Description and Performance Characteristics

The thermal control subsystem is shown schematically in Figure 1.2.12-3. It consists of bimetal-actuated louver assemblies, multilayer thermal insulation systems, and surface finishes and coatings. Spacecraft mass is used as a heat sink during transients.

The functional description of the temperature control subsystem for the spacecraft bus covers (1) internal electronic equipment, (2) G&C equipment bay, (3) propulsion subsystem, and (4) external equipment. The section on internal electronic equipment describes the major temperature control components. The external equipment section includes a description of the control method for the major pieces of remote equipment.

Internal Equipment -- The dominant temperature control factor is the rejection to space of the heat generated by internal equipment. Steady-state temperature control is achieved by high-emittance radiators rejecting heat through space-facing louvers. The influence of external environments is minimized by thermal shields and insulation. Power dissipation gradients between components on a radiator plate are reduced by conductive and radiative coupling. Equipment thermal mass reduces the effects of transients encountered during launch, maneuvers, solar occultations, and power variations.

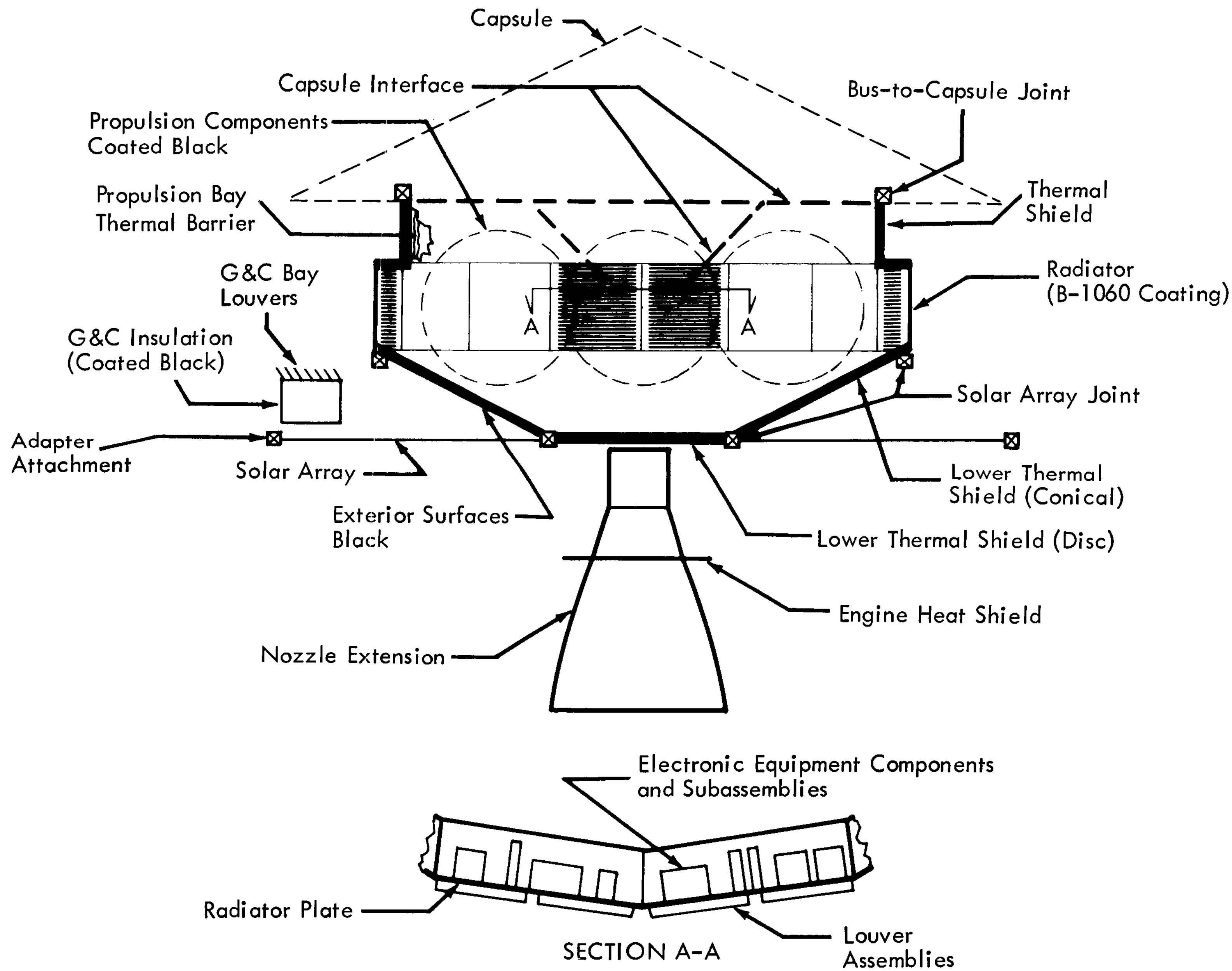


Figure 1.2.12-3: THERMAL CONTROL SUBSYSTEM

1) Louver-Radiator Assembly -- The electronic and electrical subsystems are located in four bays around the periphery of the bus. Identical louver assemblies 35.6 by 61 cm (14 by 24 inches) having an effective area of 0.202 square meters (2.17 square feet) are located over the radiator plates as required. The radiator plates are fabricated of 6061 aluminum and, except for special cases, have an effective thickness of 0.1905-cm (0.075 inch). The design provides temperature leveling across a particular assembly, and a good path for coupling to neighboring components.

The concentrated heat load of the radio power amplifier (TWTA) and the spacecraft batteries requires the use of a heavier, 0.635-cm (0.25 inch) thick, radiator plate. The TWTA assemblies radiate directly to space through cutouts in the radiator plate.

The louver assemblies are attached to structural rings above and below the electronic assemblies. The entire louver assembly frame has a low-emittance finish. Each louver blade is supported and stabilized by bearings at both ends and at the center actuator housing as shown in Figure 1.2.12-4. Each is polished inside and outside ($\varepsilon_{IR} < 0.07$, $\alpha_s/\varepsilon_{IR} < 5$) and is individually actuated thermally by a bimetal spring. These bimetal actuators are isolated from all thermal inputs, except the radiator, by seven layers of 1/4-mil aluminized mylar. The actuators are located in the actuator housing at the louver blade center bearing. The actuator can be adjusted during testing.

The louvers provide control of radiator temperature within the range of 280-297°K (45-75°F). Control is achieved by varying the effective emittance to space through a blade-opening angle between 0 and 90 degrees. Resulting performance characteristics of the assembly near Earth and near Mars is presented by Figure 1.2.12-5. Choosing a conservative 20% louver blade failure (closed position) reduces the maximum design dissipation to 25.7 watts/ft^2.

The louver assemblies are designed to operate linearly from full-closed to full-open for a radiator temperature range of 280 to 297°K (45-75°F), except for the louvers over the batteries which operate over a temperature control range of 269 to 286°K (25-55°F). This corresponds to a heat dissipation range of 3.2 to 32.6 watt/ft^2 for the equipment bay and 2.8 to 27 watts/ft^2 for the battery bay (effective emittance of 0.10 to 0.795). Figures 1.2.12-6 and 1.2.12-7 present the heat dissipation capability of an equipment and a battery louver-radiator assembly as installed on the spacecraft.

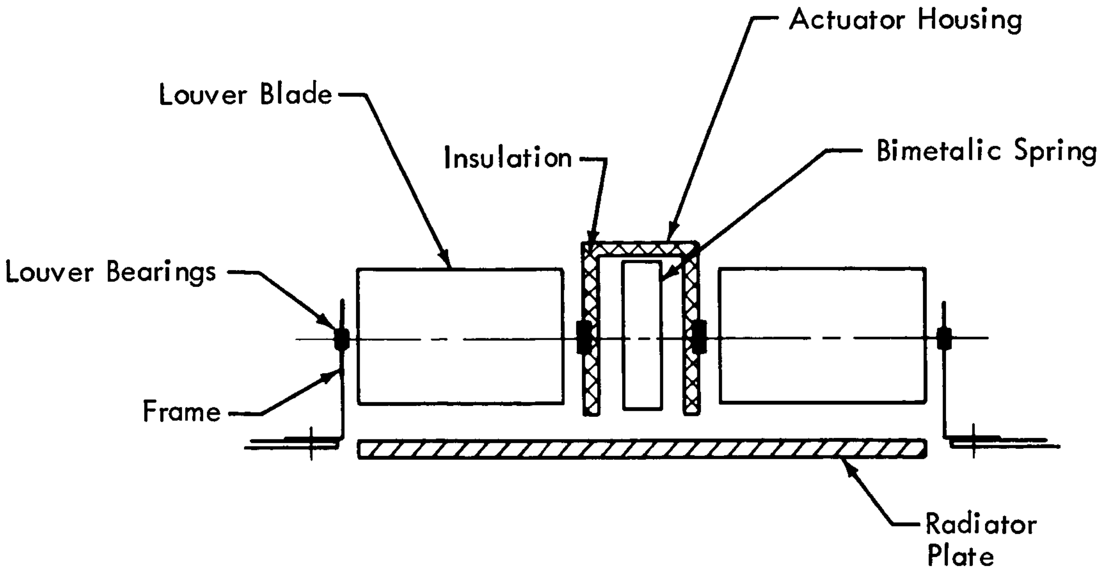


Figure 1.2.12-4: RADIATOR-LOUVER ASSEMBLY SCHEMATIC

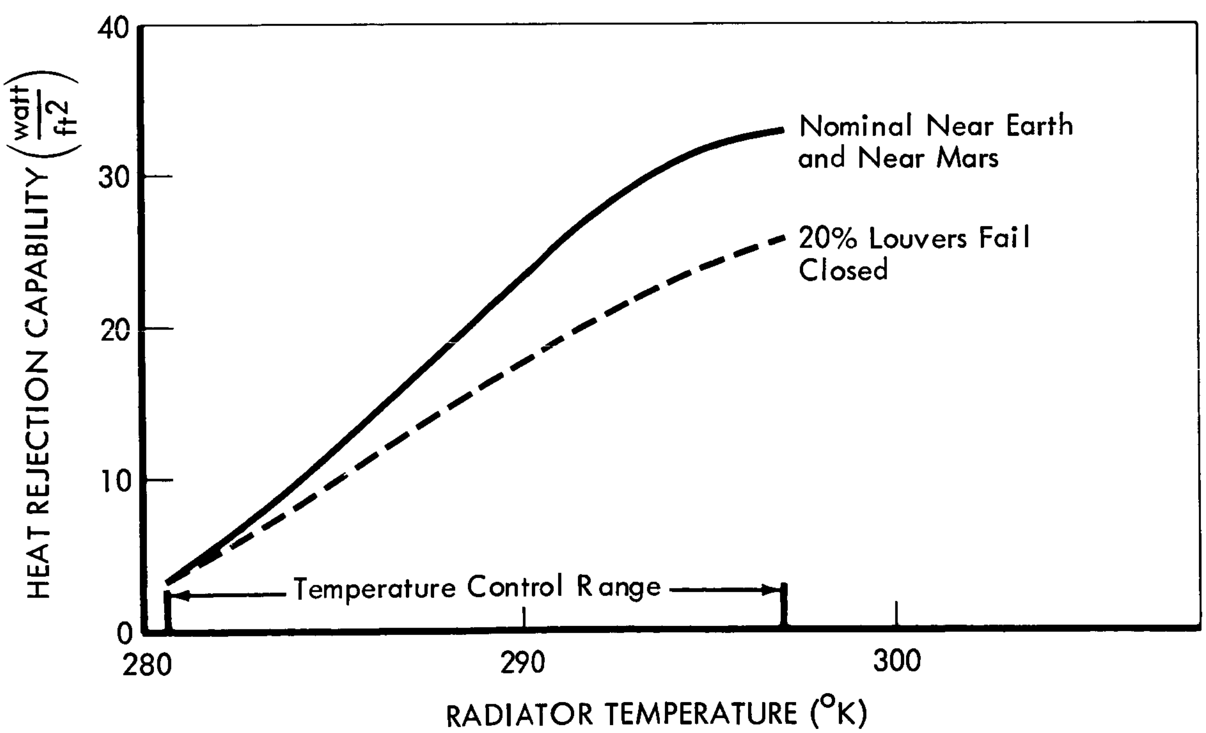


Figure 1.2.12-5: RADIATOR-LOUVER PERFORMANCE

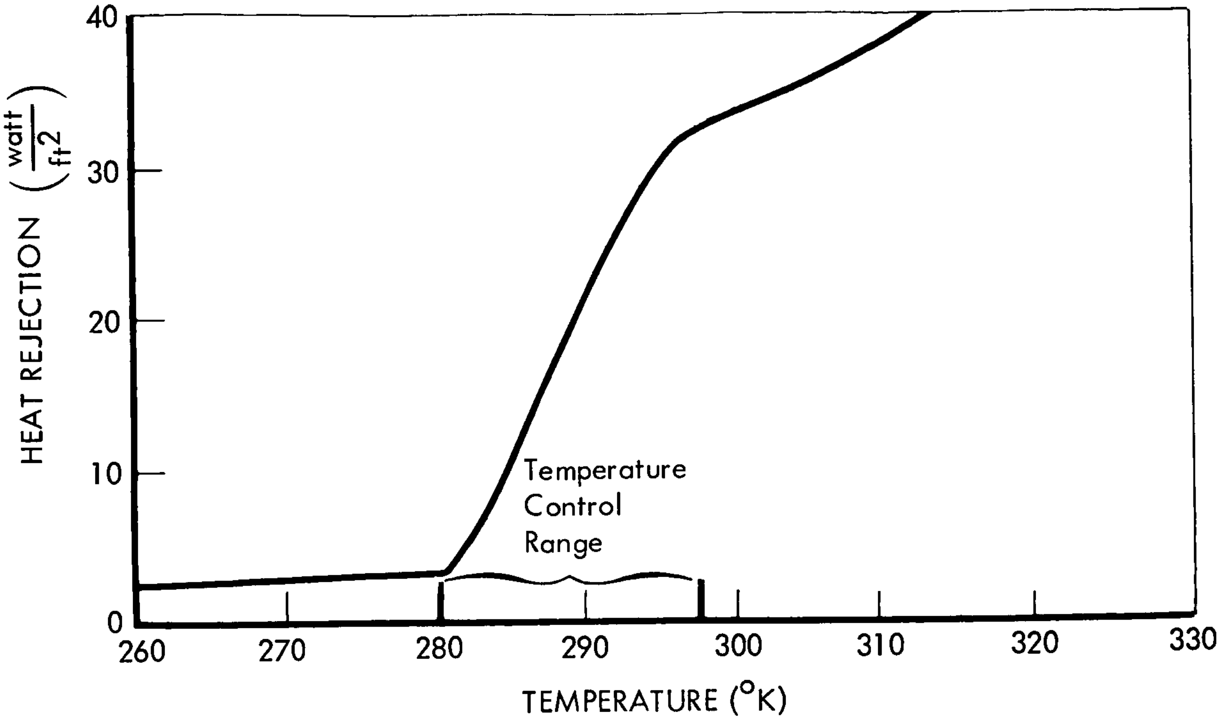


Figure 1.2.12-6: EQUIPMENT RADIATOR-LOUVER ASSEMBLY

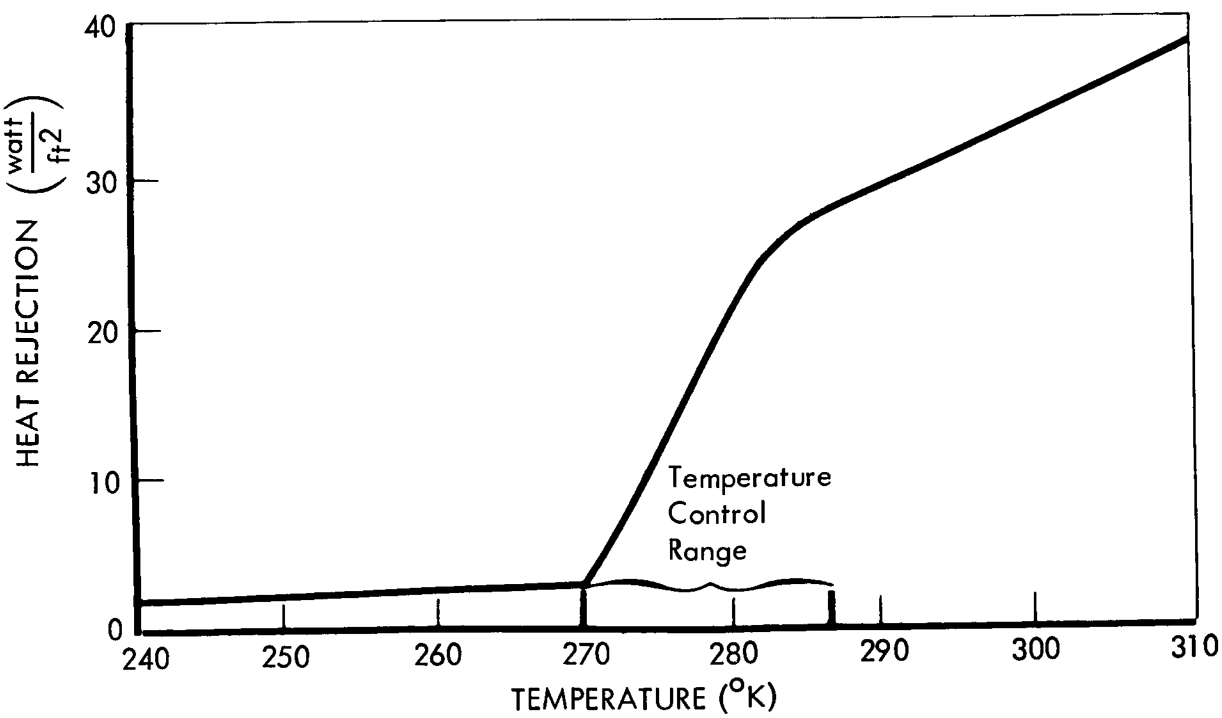


Figure 1.2.12-7: BATTERY RADIATOR-LOUVER ASSEMBLY

The maximum worst-case solar absorptance (α_s) of the louver assembly (during a maneuver) as a function of blade angle is shown by Figure 1.2.12-8. At a 90 degree blade angle, the solar absorptance (α_s) is 0.29. Figure 1.2.12-9 shows that for a maneuver where the radio and data storage bay faces the Sun for 60 minutes, the maximum data recorder temperatures will be 300°K (80°F) and the maximum TWTA temperature at the radiator will be 323°K (122°F).

2) Thermal Shield and Insulation -- Fifteen layers of 1/4-mil aluminized mylar insulation (aluminized on both sides) is installed on the primary structure. The outside space facing layer is 2-mil H-film, aluminized on one side, with the uncoated side facing space.

The capsule-to-spacecraft thermal shield is the lower capsule sterilization canister. The thermal conductance of the sterilization canister must be less than 0.90×10^{-2} watt/m^2°K (.465 x 10^{-3} watts/ft^2 °R).

The bus-to-propulsion module thermal barrier (shell liner) has a dual purpose. It isolates the bus from the propulsion module, and provides a highly reflective surface to assist in radiatively coupling the propulsion module components. This thermal barrier is comprised of six layers of 1/4-mil aluminized mylar insulation.

Aluminized mylar multilayer insulation has been tested as applied on Lunar Orbiter and in samples. These tests, combined with theoretical interpolation, have established the performance characteristics for Lunar Orbiter insulation blankets as shown in Figure 1.2.12-10. The selected insulation design requirements based on crinkled mylar blankets are shown in Figure 1.2.12-11 as a function of the penetration conductances. The sensitivity of the spacecraft thermal balance to insulation conductance is shown in Figure 1.2.12-12. The insulated wall rate of heat loss is 10% of the total spacecraft heat rejection rate when the overall wall conductance is less than 1.93×10^{-2} watt/m^2°K (1.0×10^{-3} watts/ft^2 °R).

3) Coatings and Finishes -- The space-facing surface of the radiator plate is coated with thermally white, zinc, oxide-methyl silicone coating (Boeing B-1060), which has an initial nominal solar absorptance of 0.19 and an infrared emittance of 0.89. This coating provides good heat dissipation capability during normal operation, and minimizes the solar heat input to the radiators when the spacecraft is maneuvered in such a way that the Sun impinges on the radiators.

All inner and outer surfaces of the equipment packages, and the bus internal structure, are coated with Cat-a-lac black (nominal infrared emittance of 0.9). This high-emittance coating provides maximum radiative coupling between all bus internal masses.

Table 1.2.12-2 presents the characteristics of selected coatings and finishes. Indicated tolerances are used for providing adequate performance margin.

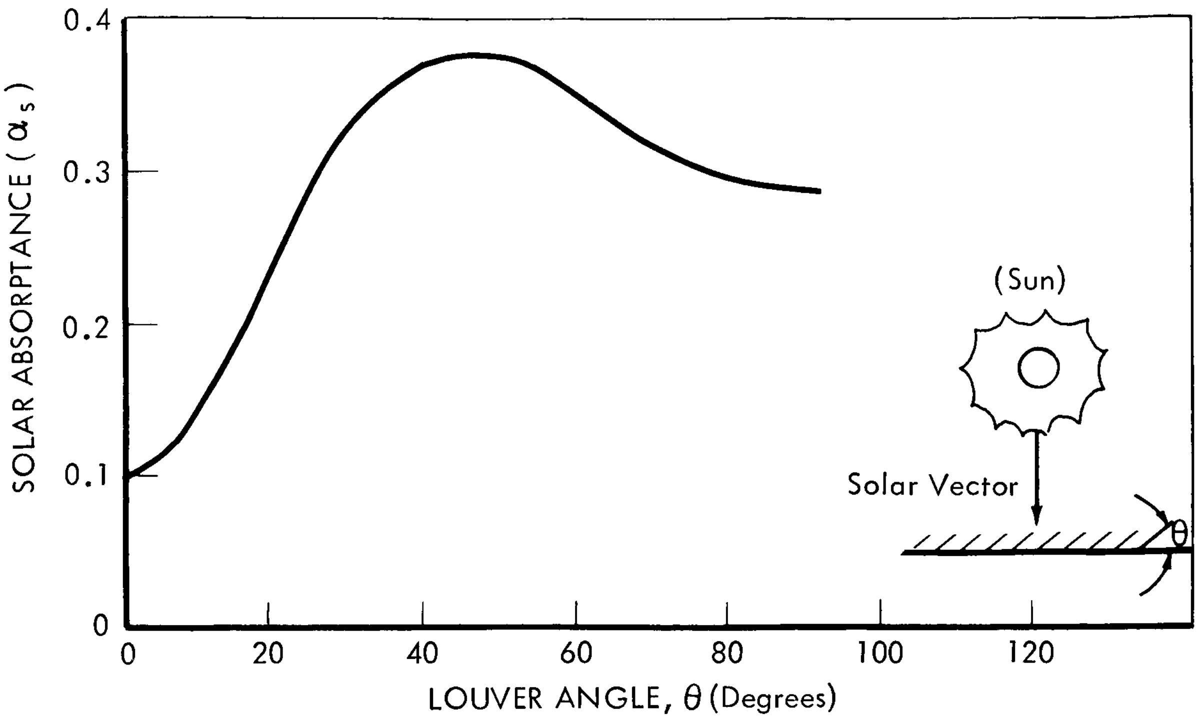


Figure 1.2.12-8: SOLAR ABSORPTANCE OF RADIATOR-LOUVER ASSEMBLY

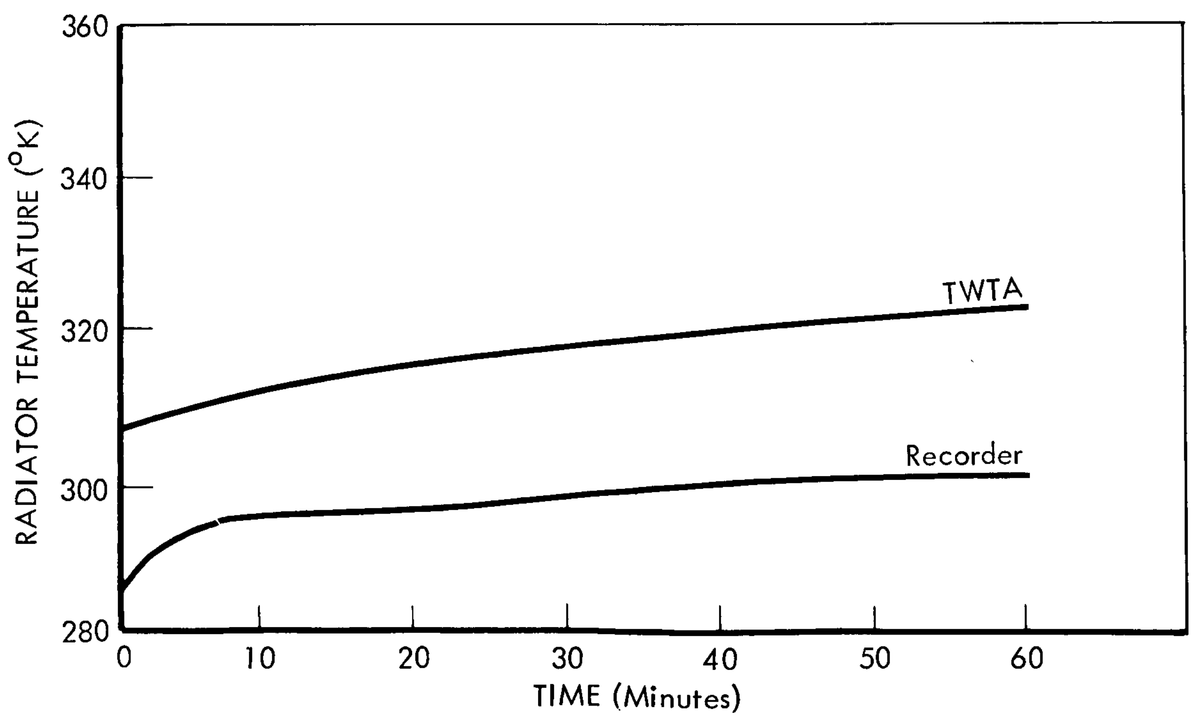


Figure 1.2.12-9: TRANSIENT RADIO/DATA STORAGE TEMPERATURES

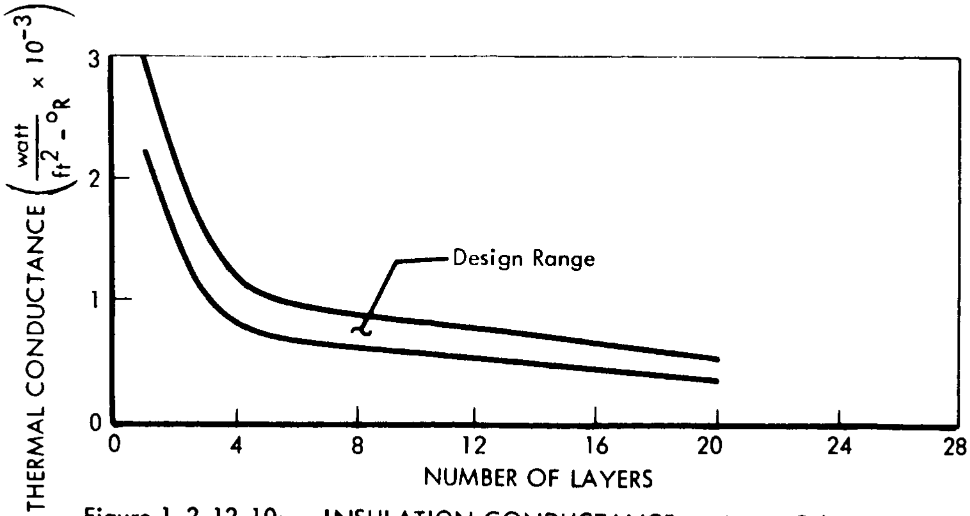


Figure 1.2.12-10: INSULATION CONDUCTANCE — Lunar Orbiter Data

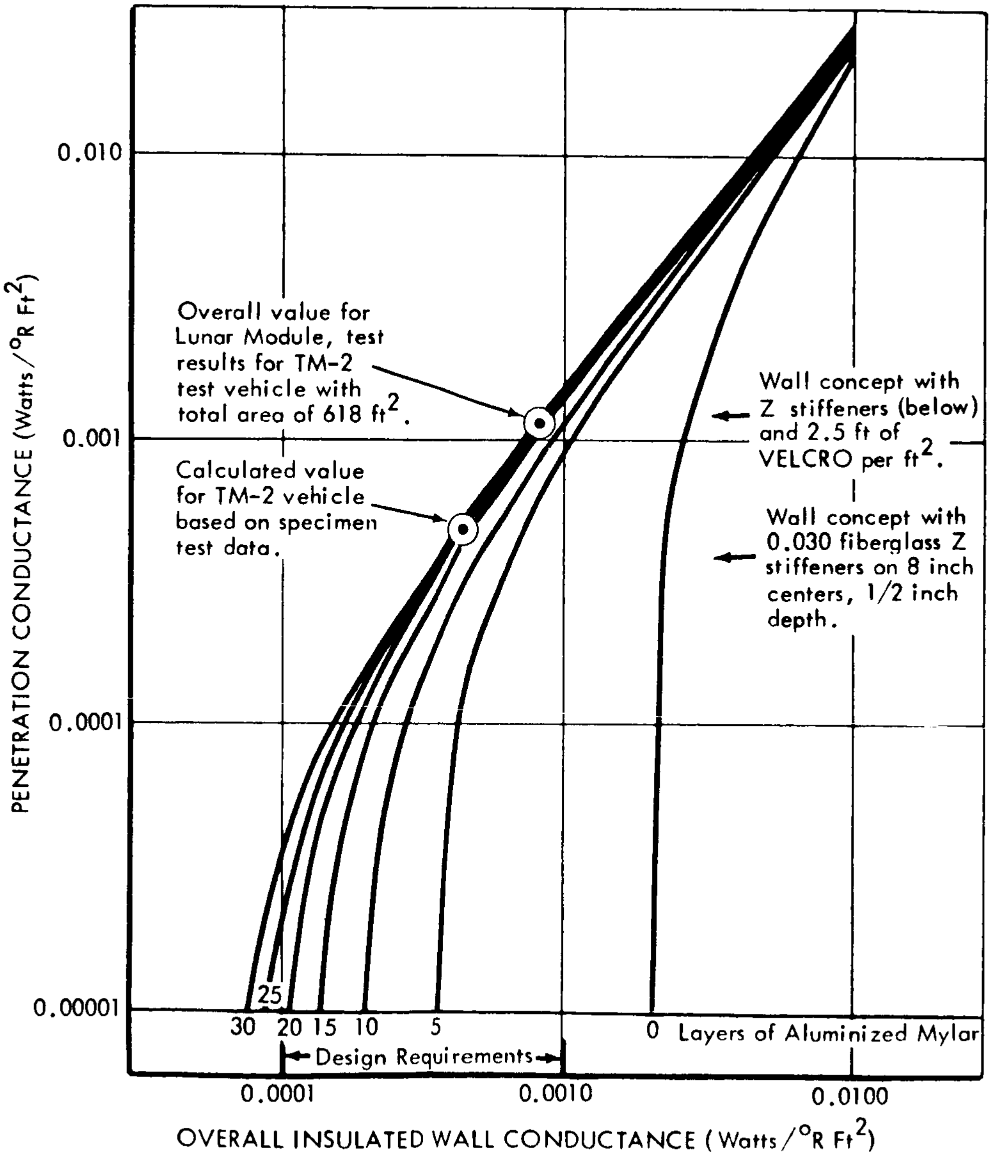


Figure 1.2.12-11: INSULATION DESIGN REQUIREMENTS

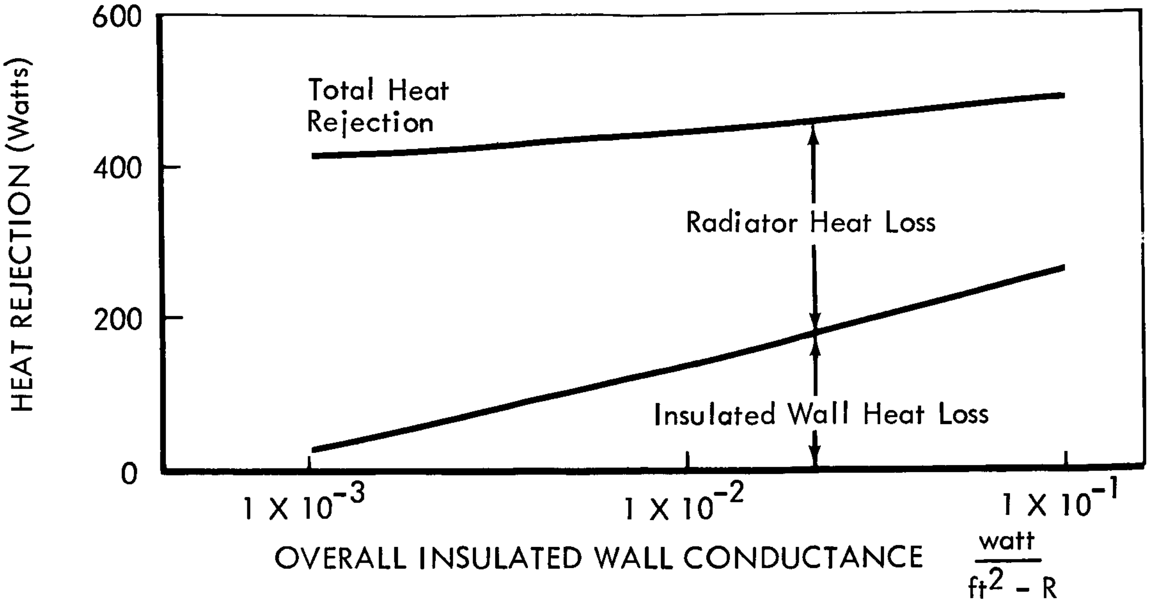


Figure 1.2.12-12: HEAT BALANCE SENSITIVITY TO INSULATION CONDUCTANCE

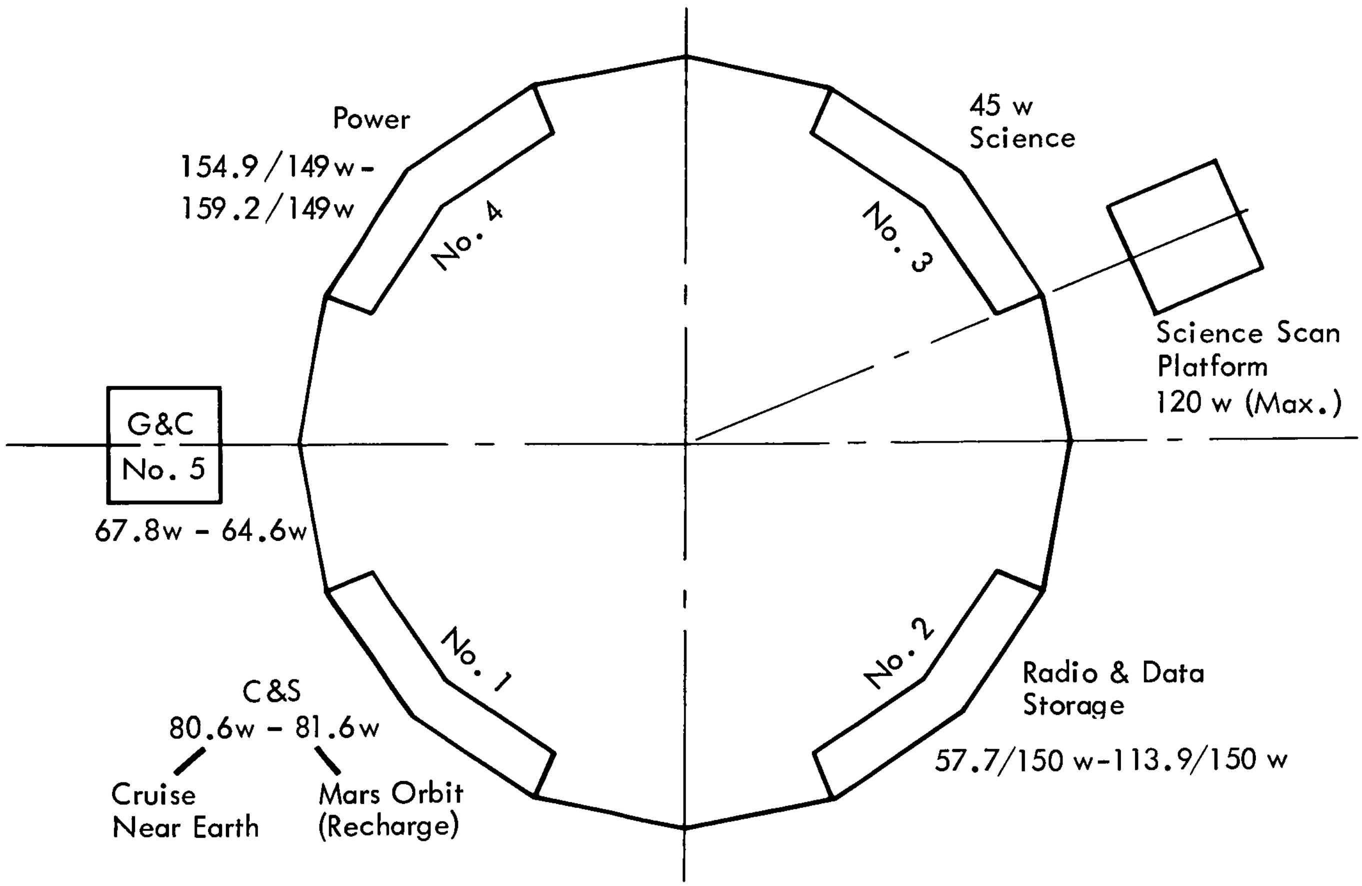


Figure 1.2.12-13: LOUVER/EQUIPMENT LOCATIONS — Heat Loads in Watts

Table 1.2.12-2: CHARACTERISTICS - THERMAL COATINGS AND FINISHES

Coating or Finish	Initial Solar Absorptivity	Emissivity
Polished Aluminum (6061 light sand blast)	0.19 ± 0.06	0.06 ± 0.03
(1100 series)	0.13 ± 0.03	0.05 ± 0.02
Vacuum Deposited Aluminum	$0.13 {+0.03 \atop -0.02}$	$0.05 {+0.03 \atop -0.02}$
Zinc Oxide-Methyl Silicone (B-1060) 10-mil	0.19 ± 0.03	0.86 ± 0.02
Cat-a-lac Black	0.95 ± 0.01	0.88 ± 0.02

4) Structures -- The primary bus structure is virtually isolated from the bus secondary structure (such as solar panel supports), and also from the propulsion module and the external environment. With this isolation, the primary structure, which is closely coupled to the electronics, provides a secondary thermal mass for temperature damping during transients. The bus structure associated with the equipment bays is coupled thermally to the electronic equipment and weighs approximately 85.6 kg (189 pounds). The effective thermal capacitance of this structure is about

$$19.9 \frac{\text{watt-hr}}{°K} \quad (37.8 \frac{\text{Btu}}{°F}).$$

An overall thermal balance of the bus internal equipment indicates a gross louver area requirement of 1.72 square meters (18.5 square feet). Because of local requirements for each assembly, particularly for the varying heat dissipation assemblies, the louver area must be greater than that indicated by a gross heat balance. Increasing the louver area substantially above that required for a gross balance tends to decrease the temperature control system performance margin. Increased louver area causes an increased heat leak, and decreases control capability of the louvers. However, because of electronic assembly thermal constraints and the desire to provide only one louver size, 2.23 square meters (24 square feet) of louvers (11 identical assemblies) are used. Where necessary, this effective louver area is readily reduced by blocking louvers (closed), or by insulating a radiator plate from space. Figure 1.2.12-13 is a schematic of the equipment arrangement, showing the relative location of the louvers. It is apparent from this diagram that the radio and power subsystems establish the standard louver radiator size.

Table 1.2.12-3 presents average temperatures of the assemblies for several mission phases. Table 1.2.12-4 presents the interface heat leak for the bus internal equipment.

Thermal capacitance of the components, assemblies, and the primary bus structure limits transient effects during ascent, Earth orbit park, and off-Sun maneuvers. Temperature profiles during maneuvers were presented earlier in Figure 1.2.12-8. The thermal control design does not constrain required spacecraft maneuvers.

Table 1.2.12-3: INTERNAL EQUIPMENT ASSEMBLY TEMPERATURES

ASSEMBLY	TEMPERATURES °K (°F)			
	NEAR EARTH	MARS ORBIT	1 HOUR ON SUN	SOLAR OCCULTATION
Guidance and Control	286 (54)	283 (49)	306 (90)	
Transponder	289 (60)	289 (61)	304 (87)	
TWTA	299 (78)	301 (81)	323 (122)	
Data Storage	285 (53)	287 (57)	304 (87)	
Command Detector	287 (57)	290 (62)	301 (82)	
Command Decoder		286 (54)		
Pyrotechnics		284 (52)		
Propulsion Electronics		286 (55)		
Computer and Sequencer		297 (75)		
Battery	271 (27)	271 (27)		280 (44)
Battery Charger	283 (50)	283 (50)		305 (89)
Inverters	294 (69)	294 (69)		294 (69)
Voltage Regulators	293 (68)	293 (68)		293 (68)
Diode Logic	279 (42)	279 (42)		280 (44)
Total Heat Dissipation (watts)	506	664		
Less TWTA	356			
Less Battery Charger		537		

Table 1.2.12-4: INTERNAL EQUIPMENT THERMAL BALANCE HEAT LEAKS

SOURCE	HEAT (Watts)		
	NEAR EARTH	MARS ORBIT	SOLAR OCCULATION
Propulsion Module Attachment	- 2.5	-2.0	-2.0
Solar Panel Struts	24	-15	-80
Propulsion Module	16	8	8
Insulation System	-20	-24	-24
Capsule Interface	-24	-24	-24
Total Heat Leak	-6.5	-57	-122
Equipment Load	486	549	549
Total Heat Load	479.5	492	427

(Louver Performance = 75 to 700 Watts)

Guidance and Control Bay -- The guidance and control subsystem is located in an isolated compartment near the edge of the solar array. The thermal balance is achieved by isolating the compartment from its environment and controlling the heat rejection to space by use of a louver assembly. Power dissipation gradients between components within the G&C package are reduced by conductive and radiative coupling. Equipment thermal mass is used to reduce the effects of transients encountered during launch, off-Sun maneuvers, and power variations. The guidance and control bay has a single radiator plate over which is located a louver array, Figure 1.2.12-14. The radiator plate is 45.7 by 81.3 cm (18 by 32 inches) and is 0.635-cm (1/4 inch) thick, 6061 aluminum. The louver temperature control range is 283 to 300°K (50°F to 80°F).

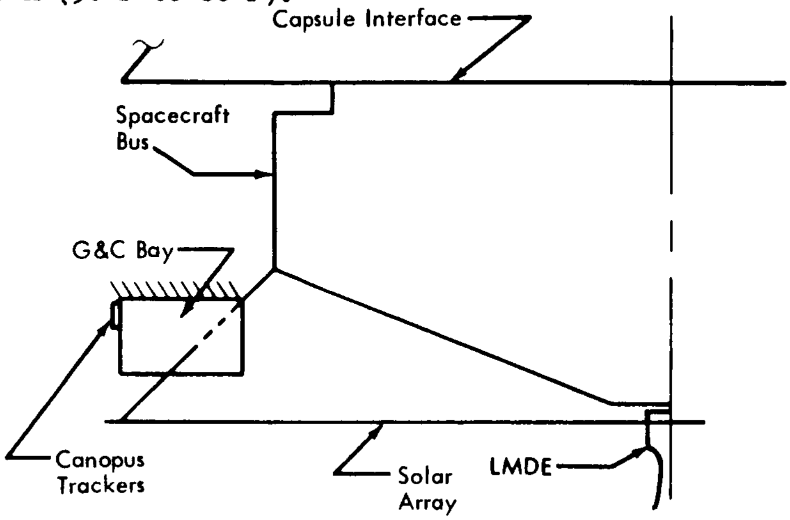


Figure 1.2.12-14: GUIDANCE AND CONTROL BAY LOCATION

Propulsion Subsystem -- Temperature control of the propulsion subsystem is achieved by proper balance of the solar heat absorbed by the LMDE, the lower thermal shield, and the heat lost to space from external surfaces and temperature-actuated louvers. Salient features of the subsystem are shown in Figure 1.2.12-3. The temperature control subsystem is designed to provide the greatest possible margin for unavoidable variations in the heat balance between the propulsion system and its environment. This is basically the same concept used for the internal equipment except that the propulsion subsystem is not slaved to a large internal power dissipation.

The primary elements of the propulsion subsystem thermal control are:

1) Thermal Mass -- The mass of the propulsion subsystem provides a thermal capacitance, including LMDE, of over 3480 watt-hr/°K (6599 Btu/°R) loaded, and 752 watt-hr/°K (1428 Btu/°R) after the orbit insertion engine burn.

2) Louvers -- Two louver assemblies, each 30.5 by 30.5 cm (12 x 12 inches) having an area of 0.09 m^2 (1.0 ft^2), are provided on the primary bus structure for temperature control of propulsion subsystem. The louver assemblies are identical to those used on the electronics except that the actuators are set for 283 to 311°K (50 to 100°F) and sense the mean temperature of the propulsion module.

3) Lower Thermal Shield -- The insulation on the sunward side of the module is comprised of two sections. The disk section (Figure 1.2.12-3) is built up of 19 layers of 1/4 mil aluminized H-film. The outside, space facing layer is 2-mil H-film aluminized on one side with the uncoated side facing space.

The conical section is also 19 layers of 1/4 mil aluminized mylar with one 2-mil layer aluminized H-film. All internal surfaces are specular ($\rho_{IR} \leq 0.95$).

4) Thermal Coatings and Finishes -- All internal tanks and plumbing are coated with the Cat-a-lac black to achieve maximum thermal coupling by direct interchange and by reflections from the specular radiation shield.

5) Insulation -- The entire propulsion subsystem is enclosed by multilayer aluminized mylar insulation. The lower heat shield, which isolates the propulsion module from the solar array, is described above. Three to six layers of 1/4-mil aluminized mylar are used to isolate the propulsion module from the electronic bays. The lower portion of the capsule sterilization canister acts as the thermal shield between the capsule and the spacecraft during capsule transport and between the spacecraft and space after capsule separation.

Analysis indicates that the temperature control of the propulsion subsystem provides adequate margins for all mission phases. A summary of propulsion subsystem temperatures is given in Tables 1.2.12-5 and 1.2.12-6.

Before launch, the propulsion subsystem is conditioned to 50°F. The total propulsion subsystem weighs 6000 kg (13,067 pounds) loaded, representing a thermal capacitance of 3480 $\frac{\text{watt-hr}}{°K}$ (6599 Btu/°R) and 225 $\frac{\text{watt-hr}}{°K}$ empty (465 Btu/°R empty). The long time constant of the propulsion subsystem, due to its thermal capacitance and insulation, is a significant compensation for the changing solar flux between Earth and Mars. A net 25-watt heat leak into the propulsion bay results in only a 0.18°K change in bulk temperature per day.

External Equipment -- The general method for maintaining the required temperature of heat-generating equipment mounted remotely to the main bus structure is to establish minimum thermal coupling to all neighboring structures and to the external environment. Exposed components are provided with individual solar shields coated with Cat-a-lac black.

Heaters are used where low temperatures could occur due to duty cycle or spacecraft maneuvering. Nonactive external equipment is controlled by use of surface finishes and coatings.

1) Gimbals for High Gain and Medium Gain Antennas and Scan Platform -- Gimbal assemblies are insulated and contain heaters. Thermal capacitance of the assembly is used to absorb intermittent power output of the motors. Each drive unit is supplied with a 3-watt heater. The heaters are turned on at the beginning of interplanetary cruise, and remain on for the entire mission. The temperature is maintained between 244 and 322°K (-20 and +120°F) throughout all mission phases.

2) High Gain and Medium Gain Antenna -- The high gain and medium gain antennas are coated on both sides with Cat-a-lac black. Diffuse black surfaces are used wherever possible on the exterior of the spacecraft because of their highly predictable thermal characteristics. The medium gain antenna average

Table 1.2.12-5: PROPULSION SUBSYSTEM TEMPERATURES

COMPONENTS	NEAR EARTH		MARS ORBIT (in Sun)		ORBIT INSERTION MAXIMUM	
	°K	°F	°K	°F	°K	°F
Propellant Tanks	308	94	297	74	—	—
LMDE						
●Combustion Chamber Internal Wall	464	376	352	175	908	1175
●Combustion Chamber Outside Insulation	412	282	315	108	577	578
●Exit Cone Internal Surface	403	265	305	89	—	—
●Nozzle Extension Internal Surface	276	37	212	-78	1230	1755
●Nozzle Extension External Insulation	269	25	210	-82	674	762
●Injector Head Internal Surface	455	359	344	159	—	—

Table 1.2.12-6: PROPULSION BAY THERMAL BALANCE HEAT LEAKS

HEAT TRANSFER MECHANISM	HEAT LEAKS (Watts)	
	NEAR EARTH	MARS ORBIT (in Sun)
Lower Thermal Barrier	-1	-8
LMDE	150	35
Capsule Interface	-2	-19
Thermal Barrier	-27	-24
Electronics	-2	2
TOTAL	118	-14

temperature is 312°K (102°F) at Earth and 262°K (12°F) at Mars encounter. The average high gain antenna temperature is 328°K (130°F) at Earth and 239°K (-40°F) at Mars. Temperature during solar occultation drops to 105.5°K (-270°F).

3) Booms, Solar Panels, and Exterior Surface -- All exterior surfaces of the bus, including booms, are coated with Cat-a-lac black except radiator plates.

The steady-state temperature profiles for the solar panels are presented in Figure 1.2.12-15. The panel temperature varies from 389°K (240°F) to 341°K (153°F) at Earth and from 300°K (80°F) to 264°K (16°F) at Mars, with a maximum radial temperature gradient of 48°K at Earth and 36°K at Mars from root to tip. The minimum temperature at end of 1.5 hour occultation is 120°K (-243°F).

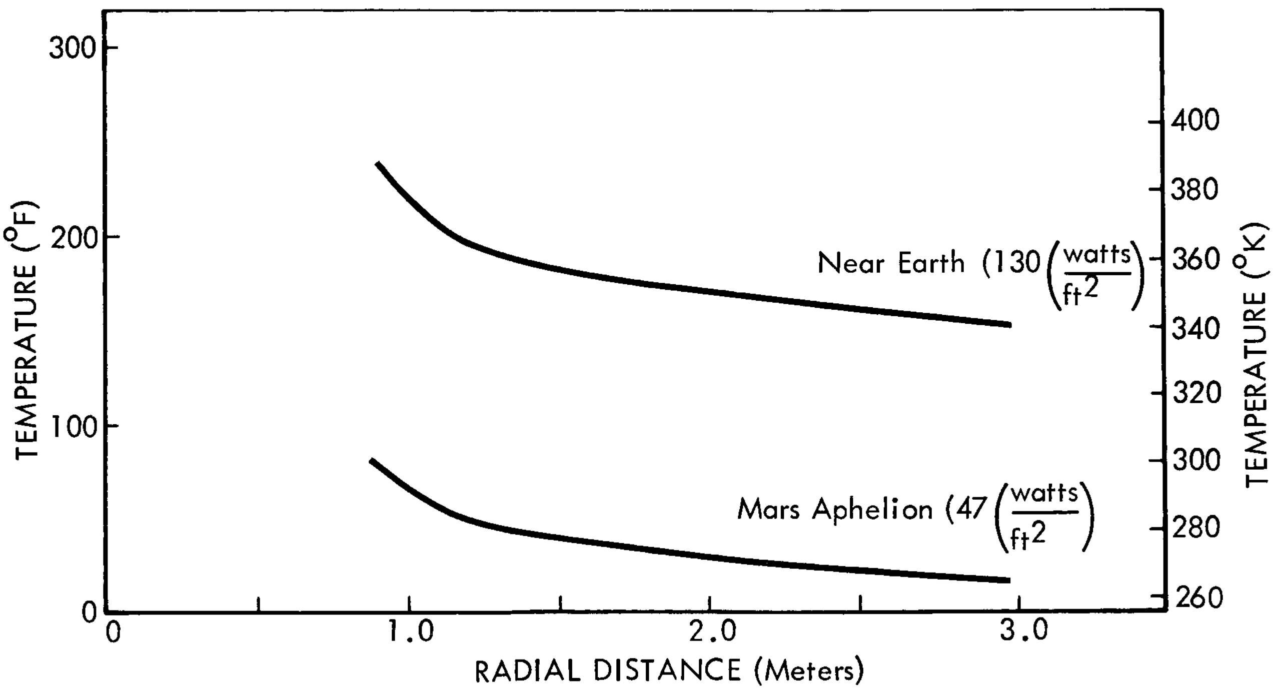


Figure 1.2.12-15: SOLAR ARRAY TEMPERATURES

1.2.12.3 Physical Characteristics

All multilayer insulation barrier assemblies are kept to a size no greater than approximately 1.22 meters by 1.22 meters (4 feet by 4 feet) to facilitate fabrication, handling, assembly, and removal. Depending on the application; the assembly is either multiple layers of 1/4-mil mylar aluminized on both sides, or multiple layers of 1/4-mil mylar aluminized on both sides with an outside cover layer of 2-mil H-film aluminized on one side, or multiple layers of 1/4-mil H-film aluminized on both sides with the outside cover of 2-mil H-film aluminized on one side. All insulation systems are assembled as crinkled layers and made into blankets (no separator material). Table 1.2.12-7 presents the temperature control subsystem weights.

Table 1.2.12-7: THERMAL CONTROL SUBSYSTEM WEIGHTS

SUBSYSTEM ELEMENT	WEIGHT (Pounds)
Louver Assemblies (24 ft^2)	18
Insulation, Equipment Bays	26.9
White Paint (24 ft^2)	3.0
Black Paint, Equipment Bays	4.6
Ground Cooling	12.0
Propulsion Bay Louvers (2 ft^2)	1.5
Insulation, Propulsion	51.7
Black Paint, Propulsion Bay	15.7
Miscellaneous	21.6
LMDE Heat Shield	29.0
TOTAL	184.0

1.2.12.4 Interface Definition

The interfaces are defined in Table 1.2.12-8.

1.2.12.5 Reliability

The louvers comprise a potential failure source in the temperature control subsystem. In addition, system performance can degrade as a result of insulation and coating degradation. The reliability analysis is based on: (1) 18-blade louver assemblies in which no more than one blade in every five (and at least five blades apart) could fail in either direction and (2) degradation of insulation and coatings without complete failure. The reliability data show that the temperature control subsystem meets the reliability goal of 0.9999.

1.2.12.6 Trade Study Summary

The following trade studies were performed and the summaries are shown in the indicated figures:

Trade Study	Figure
Louver Blade Actuation Method Thermal Coupling Between Components	Figure 1.2.12-16
Thermal Coupling to Primary Structure Thermal Coupling Between Bus and Propulsion	Figure 1.2.12-17
Solar Array Location Thermal Sensitivity	Figure 1.2.12-18
Louver Blade ℄ of Rotation LMDE Location	Figure 1.2.12-19

Table 1.2.12-8: TEMPERATURE CONTROL SUBSYSTEM INTERFACES (Sheet 1 of 2)

ITEM NO.	TYPE OF INTERFACE	INTERFACE DESCRIPTION	INTERFACING SUBSYSTEM	INPUT TO	INPUT FROM	BOUNDARY DEFINITION
1	Thermal/Functional	Control temperature of the Bus subsystems throughout all ground test and mission phases.	All	X		At the interfaces of each subsystem, control the conducted, radiated and convected heat flux.
2	Thermal/Functional	For all ground test and mission phases, specify required temperature of equipment and structure.	All		X	Temperature control specification for each subsystem.
3	Thermal	For all ground test and mission phases, define location and rate of equipment heat generation.	All		X	Definition of equipment heat generation for each subsystem.
4	Thermal/Mechanical	Satisfy thermal design requirements at Bus-to-planetary Vehicle interfaces in accordance with Temperature Control Subsystem specification.	All		X	Coatings and finishes, insulation, radiation shields, and subsystem configuration control.
5	Thermal/Mechanical	At the interfaces between all subsystems and the space thermal radiation environment, satisfy the thermal design requirements in accordance with temperature control subsystem specifications.	All		X	Coatings and finishes, insulation, radiation shields, subsystem configuration control & elec. heaters.
6	Thermal/Mechanical	For all non-faying surfaces throughout the bus interior (components, modules, subassemblies, assemblies, cables, structure, etc.), provide coatings and finishes which effect maximum feasible radiation coupling. Cat-a-lac Black shall be used where feasible.	All		X	Surface of all items include the bus (surfaces not viewing space or sun).
7	Thermal/Mechanical	Satisfy thermal design requirements at Bus-to-Planetary Vehicle Adapter interface in accordance with Temperature Control Subsystem specifications.	Structural & Mechanical, Power, Radio, Guidance & Control, Science, Pyrotechnic, Cabling & Packaging, Telemetry		X	Coatings and finishes, insulation, radiation shields, and configuration control.
8	Thermal/Mechanical	Coat all external structure surfaces with Cat-a-lac Black	Structural & Mech., Power		X	Space- or Sun-viewing structure surfaces including solar panel.
9	Thermal/Mechanical	For all faying surfaces throughout the bus, except controlled insulated joints, provide flatness and finishes which effect maximum feasible thermal conduction coupling.	All		X	All faying surfaces except those which are insulated.
10	Thermal/Mechanical	Provide bolt torques and spacing (bolts and screws) which satisfy Temperature Control Subsystem requirements for joint thermal conductance.	All		X	All bolted joints.
11	Thermal/Mechanical	Coat the space-facing side (non-faying surfaces) of radiators with B-1060 white thermal control coating.	All except Structural & Mechanical		X	All radiators.
12	Thermal/Mechanical	Provide heat leak control at all penetrations of the Bus thermal barriers and heat shields in accordance with Temperature Control Subsystem specifications.	All		X	Coatings and finishes, insulation radiation shields, and subsystem configuration control
13	Mechanical	Provide air duct for ground cooling air.	Launch Vehicle		X	All duct joint at Bus.

Table 1.2.12-8: TEMPERATURE CONTROL SUBSYSTEM INTERFACES (Sheet 2 of 2)

ITEM NO.	TYPE OF INTERFACE	INTERFACE DESCRIPTION	INTERFACING SUBSYSTEM	INPUT TO	INPUT FROM	BOUNDARY DEFINITION
14	Thermal	Sterilization canister thermal conductance.	Capsule	X		Conductance shall be less than 0.90×10^{-2} watt/m^2-°K.
15	Thermal/Mechanical	Provide air flow for temperature, pressure and dewpoint control.	Launch complex equipment		X	Bus temperature control subsystem air duct inlet.
16	Thermal/Mechanical/ Electric	Provide monitor of cooling air: a) Air mass flow rate monitoring b) Inlet air temperature monitoring c) Exit air temperature monitoring d) Air inlet dewpoint monitoring	Launch complex equipment		X	Bus temperature control subsystem air inlet and exit locations.
17	Mechanical	Provide mounitng surfaces and hole provisions for louvers.	Structural & mechanical		X	Match hole patterns to bus structure.
18	Mechanical	Provide mounting surfaces and hole provisions for bus air duct.	Structural & mechanical		X	Match hole patterns to bus structure.
19	Mechanical	Installation provisions for inner and outer bus thermal barriers.	Structural & mechanical		X	All areas (except louvers and TWTA) over inside and outside of bus.
20	Electrical/Mechanical	Provide temperature transducer installations in accordance with temperature control subsystem specs.	All		X	Temperature transducers installed at miscellaneous locations on bus.
21	Electrical	Bus temperature control subsystem engineering and operational measurement signals.	Telemetry		X	Cable connectors at telemetry subsystems.
22	Mechanical	Provide support for measurement lines and power cables.	Structural & mech. Cabling & packaging		X	Mounting provisions at miscellaneous locations.
23	Electrical/Electronic	Bus temperature control subsystem backup heater commands.	C&S		X	Cable connectors at **C & S**
24	Electrical/Mechanical	Provide emergency electric heater installations in accordance with temperature control subsystem specifications.	All		X	Cable connectors at subsystem interfaces.
25	Thermal/Mechanical	For all ground test and mission phases, define propellant mass flow rate ranges and time durations.	Propulsion		X	Definition of amplitude and time-durations of propellant flows.
26	Thermal	For each burn sequence, define the resulting transient temperatures of nozzles, exit cones, and propellants.	Propulsion		X	Definition of engine transient temperatures caused by each burn sequence.
27	Thermal/Mechanical	Satisfy Temperature Control Subsystem design requirements at the following interfaces: a) Between each subsystem and Planetary Vehicle b) Between each subsystem and space thermal radiation environment c) Between propulsion subsystems.	All		X	Coatings and finishes, insulation, radiation shields, configuration control, electric heater installations.

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER AND TITLE LOUVER BLADE ACTUATION METHOD		SELECTION
		MATRIX OF DESIGN APPROACH		
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS		Bimetallic Sensing and Actuation	Fluid Sensing and Actuation	
Performance Blade angle will vary between 0°-90° in response to temperature of radiator plate over a 30°F range adjustable from 40-120°F. Individual blade failures can be expected. Power No power will be required for actuation.	Louver blade angle must be varied to control heat dissipating capability of equipment radiator. Parameters considered in order of priority: 1. Reliability 2. Design versatility 3. Cost 4. Control characteristics	1 Each louver blade is actuated by an individual spirally-wound thermostatic bimetal strip. Used on Mariner. DISCUSSION PRO: 1) Greater simplicity than (2). 2) Demonstrated reliability on long-term missions. 3) Inherent redundancy with individual actuators 4) Adjustable temperature setting. 5) Lower weight, volume and cost than (2). CON: 1) Low gain ratio compared with (2). Full travel/25°F.	2 Multiple blades are actuated by a liquid-to-vapor expandion device. Used on Nimbus. DISCUSSION PRO: 1) Remote sensing capability is available. 2) Higher gain (full travel 17°F) possible than (1). CON: 1) Greater mechanical complexity than (1) 2) Risk of possible leakage. 3) Single actuator failure affects several blades. 4) Weight and volume penalty compared with (1). 5) Difficult temperature adjustment.	Reliability (1)(2) Design Versatility (1)(2) Cost (1)(2) Control Characteristics (2)(1) Selected Approach Bimetallic sensing and actuation (1)

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER AND TITLE THERMAL COUPLING BETWEEN ELECTRONICS COMPONENTS & SUBASSYS		SELECTION
		MATRIX OF DESIGN APPROACH		
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS		Coupled	Isolated	
Temperature Limits Normal operating temperature range shall be 45-75°F. During transients due to maneuvers and equipment power variation temperatures will exceed Flight Acceptance Tests.	A means is required to provide temperature leveling during steady-state conditions. Thermal capacitance is required to maintain temperature control during transient conditions. Parameters considered in order of priority: 1. Temperature margins 2. Reliability 3. Weight 4. Versatility	1 This approach is used by Mariner. Subassemblies are coupled with a radiator plate to provide conduction coupling to supplement radiation coupling. DISCUSSION PRO: 1) Effective thermal capacitance is improved by coupling. 2) Local hot spots are reduced. 3) Effect or power variations is reduced. 4) Louver control senses average temperature. CON: 1) Radiator weight penalty = 43.2 pounds. 2) Subassembly accessibility poor after radiator installation. 3) Temperature drop across joint = 5°F.	2 In this approach adjacent subassemblies are coupled only by radiation and conduction through mounting feet. DISCUSSION PRO: 1) Each subassembly chasses serves as radiator. 2) Good accessibility. 3) No radiator weight penalty. CON: 1) Conduction coupling through feet only. 2) Local hot or cold spots possible. 3) Louver control senses only local package. 4) Weight penalty to meet thermal inertia requirements.	Temperature Margins (1)(2) Weight (1)(2) Versatility (2)(1) Selected Approach (1) Coupled

Figure 1.2.12-16: LOUVER BLADE ACTUATION & COMPONENT THERMAL COUPLING TRADE STUDIES

<table>
<tr><td rowspan="2">TRADE STUDY SUMMARY SHEET</td><td rowspan="2">SOURCE OF REQUIREMENT</td><td colspan="3">TRADE STUDY NUMBER & TITLE THERMAL COUPLING OF EQUIPMENT ASSEMBLIES TO PRIMARY STRUCTURE</td><td rowspan="3">SELECTION</td></tr>
<tr><td colspan="3">MATRIX OF DESIGN APPROACH</td></tr>
<tr><td colspan="2">FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS</td><td>Coupled to Structure
Struct. Isolated from Space</td><td>Coupled to Structure
Struct. Coupled to Space</td><td>Isolated From Structure
Struct. Isolated From Space</td></tr>
<tr><td rowspan="2"><u>Temperature Limits</u>

Flight Acceptance Test temperatures will not be exceeded during ascent, or maneuvers.</td><td rowspan="2">Thermal Capacitance is required to maintain temperatures during transient conditions.

Parameters considered in order of priority are:
1) Temperature Margins.
2) Weight
3) Performance</td><td rowspan="2">1

Equipment is mounted directly to primary structure, structure is insulated inside and outside.

<u>DISCUSSION</u>

PRO:
1) Thermal capacitance is increased over uncoupled case.
2) No insulation between equipment and structure.
3) Structure unaffected by external heat leads.

CON:
1) Insulation weight penalty on structure inside and outside.</td><td rowspan="2">2

Equipment is mounted directly to primary structure, structure is uninsulated on outside.

<u>DISCUSSION</u>

PRO:
1) Thermal capacitance is increased over uncoupled case.
2) No insulation between equipment and structure.

CON:
1) Structure temperature affected by external heat sources.</td><td rowspan="2">3

Equipment contact with primary structure is minimized.

<u>DISCUSSION</u>

PRO:
1) Effect of structural heat leaks is minimized.

CON:
1) Only equipment mass is available during transients.
2) Insulation required between equipment and structure.</td><td>Temperature Margins (1)(2)(3)

Weight (3)(2)(1)

Performance (1)(2)(3)</td></tr>
<tr><td>Selected Approach Equipment Assemblies coupled to structure isolated from space.
(1)</td></tr>
</table>

<table>
<tr><td rowspan="2">TRADE STUDY SUMMARY SHEET</td><td rowspan="2">SOURCE OF REQUIREMENT</td><td colspan="2">TRADE STUDY NUMBER & TITLE THERMAL COUPLING BETWEEN BUS AND PROPULSION</td><td rowspan="3">SELECTION</td></tr>
<tr><td colspan="2">MATRIX OF DESIGN APPROACH</td></tr>
<tr><td colspan="2">FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS</td><td>BUS COUPLED TO PROPULSION</td><td>BUS ISOLATED FROM PROPULSION</td></tr>
<tr><td rowspan="2"><u>Temperature Limits</u>

In steady-state, the propulsion subsystem shall be maintained within 40-110°F

Functional and technical temperature limits will not be exceeded during transient conditions.

<u>Power</u>

In steady-state, no heat is generated within the propulsion module.

<u>Interface</u>

The bus shall be capable of thermal testing separate from propulsion.</td><td rowspan="2">Sufficient thermal capacitance is required to maintain temperature control during transient conditions.

The parameters considered in order of priority are:
1) Temperature margin
2) Probability of mission success.</td><td rowspan="2">1

Radiative and conductive coupling between bus and propulsion is maximized.

<u>DISCUSSION</u>

PRO:
1) Propulsion is slaved to power dissipation.
2) Thermal capacitance is maximized.
3) Uninsulated between bus and propulsion.

CON:
1) Bus and propulsion have different temperature control requirements.
2) Propulsion failures influence bus
3) Thermal testing of bus requires propulsion.
4) Propulsion coupling to external environment effects bus.</td><td rowspan="2">2

Insulation is provided between the bus and propulsion to minimize radiation and conduction coupling.

<u>DISCUSSION</u>

PRO:
1) Separate temperature control for bus and propulsion.
2) Thermal testing of bus and propulsion simplified.
3) Coupling may be provided, if required.
4) Partial failure modes rather than total.
5) Reduced heat leak for bus.

CON:
1) Weight penalty for insulation.
2) Separate louvers for propulsion required.
3) Greater temperature changes during ascent and maneuvers.</td><td>Probability of Mission Success
(2) (1)

Temperature Margins
(1) (2)</td></tr>
<tr><td>Selected Approach
(2) Bus Isolated from Propulsion</td></tr>
</table>

Figure 1.2.12-17 : EQUIPMENT/STRUCTURE AND BUS/PROPULSION THERMAL COUPLING TRADE STUDIES

<table>
<tr><td>TRADE STUDY SUMMARY SHEET</td><td>SOURCE OF REQUIREMENT</td><td colspan="2">TRADE STUDY NUMBER & TITLE SOLAR ARRAY LOCATION
MATRIX OF DESIGN APPROACH</td><td rowspan="2">SELECTION</td></tr>
<tr><td colspan="2">FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS</td><td>LMDE With Heat Shield</td><td>LMDE Without Heat Shield</td></tr>
<tr><td colspan="2">The radiant heating of external spacecraft components by the orbit insertion retropropulsion is a major consideration in the Voyager configuration.

The allowable LMDE heat flux from the engine nozzle extension is an equivalent black body at 1110°F (2971 watts/ft^2). The solar array can withstand temperatures of 300°F and transients of 4°F/sec.

The parameters considered are:

1) Mission Success
2) Weight
3) Design Versatility
4) Cost</td><td>1.
LMDE with a heat shield in the plane of the nozzle extension attach point. The shield is sufficiently large to keep the hottest point on the array below 302°F.

PRO:

1) Lower solar array temperatures
2) Shorter overall S/C length (solar array 83 inches from nozzle exit plane)

CON:

1) Added weight of heat shield
2) Heat shield attachment mechanism weight
3) High local nozzle extension temperature (adjacent to heat shield)</td><td>2.
LMDE with no heat shields for blocking the nozzle extension radiant heat flux.

PRO:

1) Simpler engine installation
2) No heat shield - lower peak nozzle extension temperatures.

CON:

1) High solar array temperatures
2) Long overall spacecraft length (solar array 113 inches from nozzle exit plane)</td><td>Mission Success
(1) (2)
Weight
(1) (2)
Design Versatility
(1) (2)
Cost
(2) (1)

SELECTED APPROACH

LMDE With Heat Shield
(1)</td></tr>
</table>

<table>
<tr><td>TRADE STUDY SUMMARY SHEET</td><td>SOURCE OF REQUIREMENT</td><td colspan="3">TRADE STUDY NUMBER & TITLE THERMAL SENSITIVITY ANALYSIS
MATRIX OF DESIGN APPROACH</td><td rowspan="2">SELECTION</td></tr>
<tr><td colspan="2">FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS</td><td>Insulation Conductance</td><td>Capsule Interface Conductance</td><td>Propulsion Bay Temperature</td></tr>
<tr><td colspan="2">Determine the sensitivity of the Voyager thermal design to:
1) insulation conductance
2) capsule interface conductance
3) propulsion bay temperature</td><td>1.
Heat loss through the spacecraft insulation reduces the louver control capability. The values of effective conductance considered are:
(a) 10^{-3} watt/ft^2-R
(b) 10^{-2} watt/ft^2-R
(c) 10^{-1} watt/ft^2-R

PRO:

1) Percent heat loss through insulation < 10% with (a).
2) Heat gain at Earth negligible.

CON:

1) Heat loss high with (b) and (c).
2) Difficult to achieve low conductance.</td><td>2.
Heat loss through the spacecraft-capsule interface (joint separation) will change considerably when the capsule is separated.

Values considered are:
(a) 4×10^{-12} watt/ft^2-R^4
(b) 4×10^{-11} watt/ft^2 - R^4
(c) 4×10^{-10} watt/ft^2-R^4
PRO:

1) Capsule interface heat loss is less than 5% (b).
2) Capsule sterilization canister remains in spacecraft.

CON:

1) Low conductance (a) difficult to achieve.</td><td>3.
Heat gain or loss to the propulsion bay can increase or decrease the louver capability.

Values considered are
(a) 570 540 °R
(b) 550 520 °R
(c) 530 500 °R

PRO:

1) Very little influence on spacecraft equipment heat balance.</td><td>Insulation
(a)

Capsule Interface
(b)

Propulsion Temp

No Influence

SELECTED APPROACH
Insulation Design Goal - 10^{-3} w/ft^2-R
Capsule Interface < 4×10^{-11} w/ft^2-R^4
Propulsion temp 570-500 °R</td></tr>
</table>

Figure 1.2.12-18: SOLAR ARRAY LOCATION & THERMAL SENSITIVITY TRADE STUDIES

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE: LOUVER BLADE ℄ OF ROTATION		SELECTION
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS		MATRIX OF DESIGN APPROACH		
		Rotation About One Edge	Rotation About Blade Centerline	
<u>TEMPERATURE LIMITS</u> Temperature of the electronics will be maintained between 45-75°F control range during steady-state conditions.	The maximum effective emittance is desired with a nearly uniform variation over the 0-90 degrees range of the louver blades. Parameters Considered: Performance Stability	1. Louver blades are rotated about the lower (nearest the radiator) edge. Blades move in opposite directions. <u>DISCUSSION</u> <u>PRO:</u> 1) 90° performance is slightly better than (2). <u>CON:</u> 1) Intermediate blade angles, the performance is lower than (2). 2) Near 90 degrees, the performance is rapidly increasing, $\frac{dQ}{dT} = \infty$ 3) More difficult to actuate with bimetallic actuators.	2. Louver blades are rotated about the centerline of the blade. <u>DISCUSSION</u> <u>PRO:</u> 1) Nearly linear performance with blade angle up to 60-70 degrees. 2) Near 90 degrees, performance is nearly constant, $\frac{dQ}{dT} = 0$ <u>CON:</u> 1) Slightly lower performance at 90 degrees than (1)	Performance (2) (1) Stability (2) (1) SELECTED APPROACH ℄ rotated louver blades (2)

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE: LMDE LOCATION		SELECTION
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS		MATRIX OF DESIGN APPROACH		
		Maximum Coupling	LMDE Isolation	
The temperatures of the LMDE during long duration exposure to the solar environment approach 400°F near Earth and 100°F near Mars. The degree of coupling of the engine to the spacecraft directly affects the spacecraft thermal control capability and the engine temperatures. Parameters Considered: Thermal control capability Reliability Engine temperatures		1. Head of LMDE located inside the spacecraft bus envelope. <u>PRO:</u> 1) Lower LMDE temperatures 2) More design flexibility <u>CON:</u> 1) High heat leak into spacecraft near Earth. 2) Large variation of spacecraft heat leak ≅ 66%. 3) Insulation of spacecraft bus more difficult.	2. Head of the engine located forward of the solar array. <u>PRO:</u> 1) Reduced heat leak into the bus during orbit insertion and steady-state. 2) Smaller variation in heat leak ≅ 40%. 3) Easy to isolate. <u>CON:</u> 1) Higher LMDE internal temperatures.	Thermal Control (2) (1) Reliability (2) (1) Temperature (1) (2) SELECTED APPROACH LMDE Isolation (2)

Figure 1.2.12-19: BLADE ROTATION & LMDE LOCATION TRADE STUDIES

1.2.12.7 New Technology and Development Items

There are no new technology or development items required for thermal control of the Voyager spacecraft.

1.2.12.8 Growth Potential

The thermal control subsystem design can accommodate substantial changes in heat loads due to the increased scope of scientific objectives in future missions.

1.2.13 Cabling and Packaging

1.2.13 Cabling and Packaging

1.2.13.1 Cabling

Design Constraints and Requirements -- The design constraints and requirements are summarized in Table 1.2.13-1.

Functional Description and Performance Characteristics -- The cabling provides the following functions within the spacecraft system:

- Interconnection of electrical power and signals between spacecraft subsystems, including propulsion module.
- Electrical connections for integration of science, capsule, and launch vehicle systems with the spacecraft.

Cabling components include the following:

- Upper cabling assembly
- Lower cabling assembly
- Power distribution
- Pyrotechnic cabling
- Propulsion module cabling
- RF coaxial cables
- Capsule connector cabling
- Launch vehicle connector cabling
- Science connection cabling
- Cable supports and clamps

The upper and lower cabling assemblies are composed of individual cables grouped according to functional requirements and electrical characteristics and formed into an assembly for common path routing and distribution within the spacecraft structure.

Figure 1.2.13-1 shows the plan view of the spacecraft and the location and configuration of the cable trays within the structure for installation and protection of the electrical cabling. The cable trays are located above and below the electronics bays and provide for peripheral routing of cables with vertical distribution of the cable terminations to each electronic bay.

The upper cable tray contains command and signal cables, the capsule separation connector cables, and other cables to peripheral components located in the upper part of the spacecraft. The lower cable tray contains the power distribution cables, pyrotechnic cables, antenna cables, and other cables to peripheral components located in the lower part of the spacecraft.

Table 1.2.13-1: CABLING DESIGN CONSTRAINTS AND REQUIREMENTS

CONSTRAINTS AND REQUIREMENTS	
GENERAL	Design based on qualified standards, materials, and processes. Individual cables shall provide point-point electrical connection between source and user. Cabling assemblies shall be comprised of individual cables grouped by functional characteristic: • Low Signal • High Signal • Power • Switching • RF Coaxial All electrical conductors shall be insulated. Cables shall be supported and clamped to prevent excessive strain, abrasion, and insulation cold flow under all assembly, test, and operating conditions. Pyrotechnic cables shall be separate cables in accordance with AFETRM 127-1.
WIRING	Wire shall be copper, stranded-silver plated, AWG #18 - #24 teflon insulation, (BMS 13-24, similar to MIL-W-16878D, Type E) Wire size selection criteria based on: • Current - Derate 50% • Voltage Drop - Source to User • Shielding - Twisted Pair - Quads Insulation - Minimum resistance 100 megohms
CONNECTORS	NAS 1599, MIL-C-26482, and MIL-C-26500 series shall be used (Selection of connectors is determined by subsystem interface) Connectors derated 50% of full load current.
ROUTING	Minimize circuit path length within structural constraints. Cables shall be routed to minimize electromagnetic interference, crosstalk, radiation, and pickup. Pyrotechnic cables shall be separately routed in accordance with AFETRM 127-1 Exposed cables shall be supported and protected to meet space environmental requirements.

Table 1.2.13-1: CABLING DESIGN CONSTRAINTS AND REQUIREMENTS (cont'd)

CONSTRAINTS AND REQUIREMENTS	
GROUNDING	Power ground shall be one point only. Grounding network shall provide minimum common circuit impedance. Conductor shields (except RF) - driven end grounded. RF shields - both ends grounded. All connector shells shall be bonded to chassis ground.
INTERFERENCE	All circuits which are capable of generating electromagnetic interference shall be shielded. All pyrotechnic cables shall be twisted pair-shielded. All power distribution cables shall be twisted pair-shielded.
POWER LOSS	Less than 1% of total electrical power demand.
FAILURE RATES	Connections (Per Hour) Welded 0.00008×10^{-6} Wire Wrap 0.000005×10^{-6} Soldered 0.0001×10^{-6} Crimped 0.0008×10^{-6} Connectors Rack & Panel 0.0009×10^{-6} Printed Circuit 0.001×10^{-6}
RELIABILITY	0.994

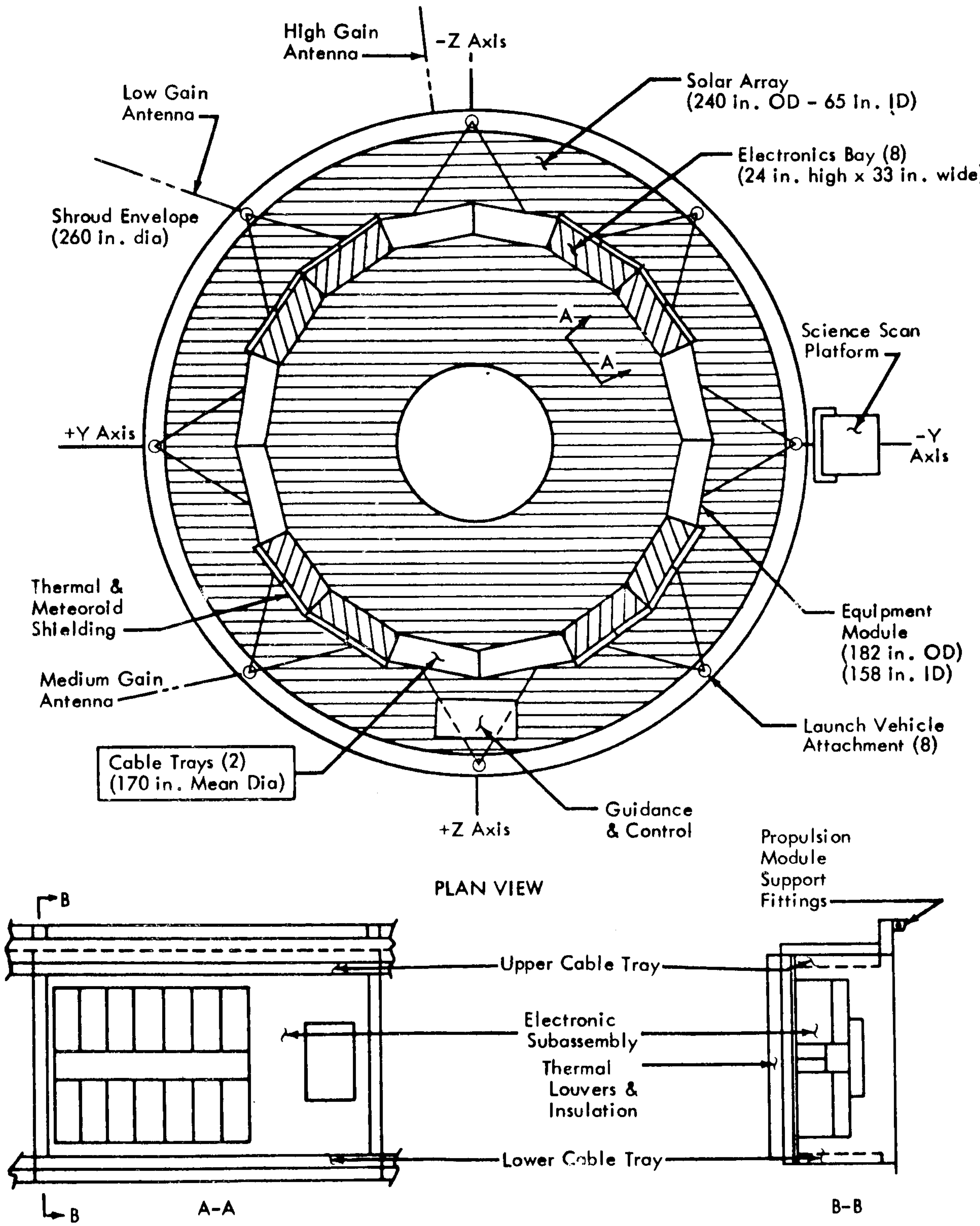


Figure 1.2.13-1: SPACECRAFT CABLING

The selection of routing path and separation of individual cables provide control of electrostatic and electromagnetic fields. Utilization of shielding, twisted pairs, and quads in individual cables provide control of pickup, radiation, crosstalk and interference within and between cable runs.

The individual cables which comprise the cabling assemblies provide point-to-point wiring between terminations and are fabricated, tested, and installed as single units. The use of individual cables provides flexibility for adjustment, maintenance, trouble analysis, and repair, and for subsequent growth and changes.

The propulsion subsystem cabling is designed to the same constraints and requirements listed in Table 1.2.13-1. The subsystem cables are routed within the propulsion module structure and attached to the truss structure for protection and support.

Physical Characteristics -- Figure 1.2.13-2 shows the temperature rise as a function of current flow for several sizes of copper wire when bundled into a 27-conductor cable. The wire size for a given load current is derated 50% and selected for an allowable temperature use of 30°C, or such limit as determined by thermal analysis of the spacecraft system and other constraints imposed by space environment, thermal protection, and insulation characteristics. Wire size smaller than AWG-24 is not used because of inadequate mechanical strength. Multiple wiring is used for circuits that exceed current rating for AWG-18. Redundant and alternate path wiring included in the cabling subsystem design where required to optimize performance, improve reliability, and to accommodate growth and changes.

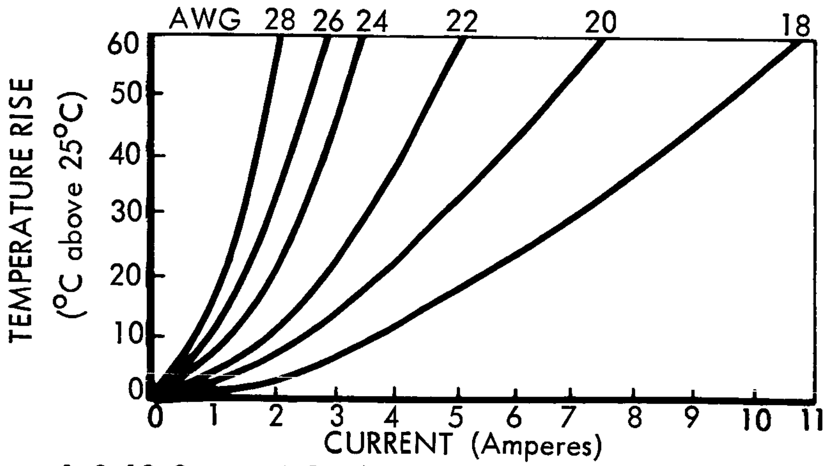


Figure 1.2.13-2: TEMPERATURE RISE — 27 CONDUCTOR CABLE

Individual cables are routed, supported, clamped, and protected by the spacecraft structure. Figure 1.2.13-3 shows the cabling concept for a typical electronics bay utilizing the upper and lower cable trays for peripheral routing between bays.

Cables that have compatible electrical characteristics have common routing. Other cables, e.g., power and pyrotechnic, are separated to comply with performance and safety requirements. All circuits that generate or are susceptible to electrical interference are shielded and routed to minimize pickup, radiation, and crosstalk.

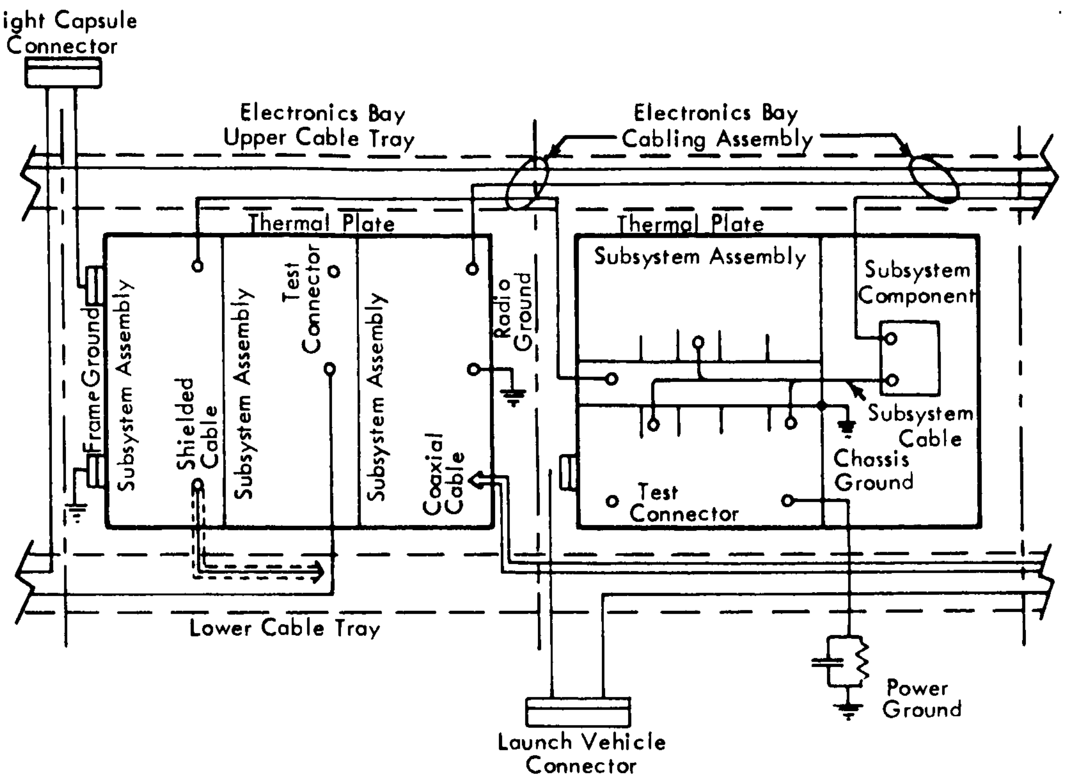


Figure 1.2.13-3: TYPICAL CABLING CONCEPT — ELECTRONICS BAY

All cables will be assembled under clean room conditions. All connectors, wire and materials will be processed to minimize entrapped contaminants that could cause excessive outgassing, lower corona breakdown, or fungus growth. Qualified materials and processes for manufacture and handling of cables for space application are available from the Lunar Orbiter program.

The power consumption and weight are as follows:

Power Consumption	1% Total Power Demand	
Weight Estimate	Spacecraft Module	240 pounds
	Propulsion Module	25 pounds
	Total	265 pounds

Interface Definition -- The cabling has electrical interfaces with all of the spacecraft subsystems, and the science, capsule, and launch vehicle systems, and has mechanical interfaces with the spacecraft structure for support, protection, and thermal control.

Reliability -- The assessed reliability for the cabling subsystem is 0.994.

Trade Study Summary -- The cabling trade studies are summarized in Figures 1.2.13-4 and 1.2.13-5.

New Technology and Development Items -- No new technology or development is required for the cabling.

Growth Potential -- Since each cable is a separate unit that is fabricated, tested, and installed on an individual basis, the cabling subsystem has inherent adaptability and flexibility to accommodate growth and changes. Individual cables can be modified, wires interchanged, or replaced as necessary to meet growth requirements.

1.2.13.2 Packaging

Design Constraints and Requirements -- The design contraints and requirements for the packaging subsystem are shown in Table 1.2.13-2.

Functional Description -- The general features of the Task D packaging system include equipment mounted internally to six 24- x 33-inch secondary structural panels and a separate guidance and control bay. A maximum height of 12 inches is allowed for assemblies, connectors, covers, and cabling.

The major elements of the Task D packaging configuration, as shown in Figure 1.2.13-6, are thermal panel, retainer assembly, module, interconnections, and nonstandard assemblies. The modules are grouped and cabled together by the retainer to form a standard assembly. Nonstandard assemblies (purchased blackboxes) and the standard assemblies mount on the thermal panel. The guidance and control electronics are packaged as a subsystem in a unique configuration that mounts outboard of the equipment and above the solar panels.

1) Thermal Panel -- The functions of the thermal panel are to provide the mounting structure for the electronic assemblies and modules, to provide a mechanism that conductively couples the heat load to the radiating surface, and to provide conductive mounting and electrical bonding for the hardware. The 24- by 33-inch thermal panel will have a thickness of 0.05 to 0.25 inch with structural stiffeners and bosses, to provide stiffness and thermal mass in the areas of the individual electronic package hardware mounts. The material of the thermal panel will be aluminum to provide adequate thermal conductance, adequate stiffness and strength, and be finished or coated to meet the thermal radiation, and electrical conductivity requirements.

   The thermal panel with its stiffeners and the structure of the standard and nonstandard assemblies will be designed to have a dynamic response of 100 Hz or less to effect isolation for the higher resonant frequencies of the electronic modules and assemblies.

2) Retainer -- The retainer consists of a rectangular frame without a bottom; its function is to assemble or group the modules to form a subsystem assembly as an entity for attachment to the thermal panel.

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE: CABLE HARNESS VERSUS INDIVIDUAL CABLES		
		MATRIX OF DESIGN APPROACH		SELECTION
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS		(A) CABLE HARNESSES	(B) INDIVIDUAL CABLES	

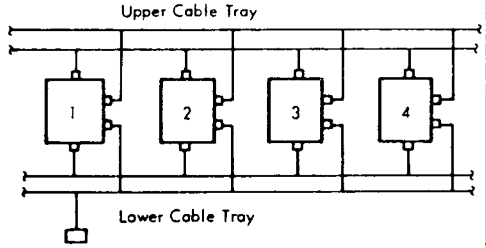
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

The cabling interfaces with all spacecraft subsystems to interconnect electrical power and signals and with the science, flight capsule, and launch vehicle systems for electrical connections with the spacecraft systems. The spacecraft structure provides physical interfaces for routing, support, protection, and thermal control.

Parameters considered are:

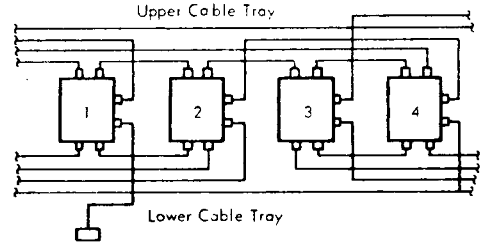
1) Design complexity
2) Fabrication and test
3) Installation and checkout
4) Performance and reliability
5) Trouble analysis and repair
6) Changes and growth
7) Weight
8) Cost

(A) CABLE HARNESSES



1) Design is unique and requires three dimensional mockup for form and fit.
2) Fabrication is more complex requiring forming boards and manual operations. Cable tests are more complex.
3) Special assembly, handling, and installation equipment is required to prevent damage to connections and joints.
4) Wire runs are fixed by cable harness configuration. Adjustments to reduce electrostatic and electromagnetic fields, crosstalk, radiation and pickup are restricted. Power loss is higher because of compromise cable harness routing requiring longer wire lengths. Power loss can only be reduced by increased wire size or additional wires.
5) Trouble analysis, repair and/or replacement is difficult. Replacement requires many disconnects.
6) Cable harnesses are not adaptable or flexible to growth or changes.
7) Integrated connectors can be used resulting in fewer subsystem connections. The heavier cable harness requires more support and larger clamps.
8) Basic cost is equivalent but change cost is greater.

(B) INDIVIDUAL CABLES



1) Design is simple requiring point-point wiring between terminations. Length is only variable.
2) Fabrication is simple and adaptable to machine methods. Cable test is simple.
3) Standard handling and installation equipment can be utilized.
4) Cable wiring assignments and separation can be adjusted to reduce electrostatic and electromagnetic fields, crosstalk radiation, and pickup between cable runs. Routing of cables can be by most direct path resulting in less power loss for given wire size.
5) Trouble can be isolated to individual cables and corrected by remove/replace with minimum of disconnects.
6) Individual cables are flexible to growth and changes.
7) More subsystem connectors are required to utilize individual cables. Connector weight is increased but wire weight is reduced, requiring less support and smaller clamps. Total weight is equivalent.
8) Basic cost is equivalent but change cost is less.

SELECTION

1-B
2-B
3-B
4-B
5-B
6-B
7-Equal
8-B

SELECTED APPROACH

Use individual cables.

Figure 1.2.13-4: CABLE TRADE STUDY

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE: SHIELD NOISE GENERATING OR SHIELD NOISE SUSCEPTIBLE CIRCUITS		SELECTION
		MATRIX OF DESIGN APPROACH		
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS		(A) GENERATING CIRCUITS	(B) NOISE SUSCEPTIBLE CIRCUITS	
It is necessary to control the electromagnetic fields, crosstalk, radiation, and pickup in the cabling which can interfere with the performance of the spacecraft, science, and flight capsule system. Parameters considered in this study are: 1) Design complexity 2) Performance 3) Reliability 4) Weight 5) Cost		1) The identification of circuits that are capable of generating electrical noise is complex because of the multiple secondary and ground loops. Major noise generating circuits such as power, solenoid switching, and pyrotechnic are obvious; however, other circuits that conduct transients of 0.6 volts/microsecond or more are difficult to identify. 2) Shielding of noise generating circuits only is not completely effective due to common ground impedance and secondary loops and does not isolate circuits from external noise sources. Reduction of interference within the spacecraft is necessary to provide proper environment for instrumentation and sensors. 3) Reduction of electrical noise generated within the spacecraft will enhance reliability of the spacecraft and the interfacing science and capsule systems. 4) Shielding weight is greater since larger wire sizes are used for power and switching circuits and the number of potential noise generating circuits is probably more. 5) Cost is increased if more circuits are shielded, but otherwise is equivalent.	1) Sensitive circuits must be shielded to control induced interference from both internal and external sources. Electromagnetic fields are present in the test and launch environment and from radiation in space. Shielding is necessary since it is not possible to control these external sources of EMI. Identification of noise susceptible circuits is relatively simple. 2) Shielding of noise sensitive circuits is necessary to meet S/N level requirements and to ensure system performance within the specified limits. 3) Shielding of noise susceptible circuits will enhance subsystem reliability and provide increased confidence of mission success. 4) Shielding weight is less if only noise-susceptible cable circuits are shielded, since signal circuits which require shielding have smaller wire size and are probably fewer in number. 5) Cost is less if fewer circuits are shielded, but otherwise is equivalent since all power distribution and pyrotechnic circuits must be shielded in either case.	1-B 2-Both required 3-Both required 4-B 5-Equal Study results above, show trade was not applicable since both methods of shielding are required to meet performance and reliability parameters for mission critical events. SELECTED APPROACH Both noise generating and noise susceptible circuits require shielding.

Figure 1.2.13-5: NOISE SHIELD TRADE STUDY

Table 1.2.13-2: PACKAGING DESIGN CONSTRAINTS AND REQUIREMENTS

	CONSTRAINTS AND REQUIREMENTS
Size & Geometry	Volume Allocation: 50 cu ft Weight Allocation: 680 pounds Total thermal radiating area required: 24 sq ft for average thermal panel temp. 60°F
Environmental	Natural: Per TM-X-53616 Induced: Per D2-82746-1
General	1) Space-qualified, off-the-shelf assemblies shall be used where available without repackaging if thermal design is compatible (direct attachment to the thermal panel). 2) In-house hardware shall be designed and fabricated in standard configurations to the maximum extent practicable. 3) New procured assemblies shall be designed to a standard configuration unless reliability and cost penalties are incurred. 4) The standard configuration will be recommended for use for the GFE Science Package. 5) The structural tiedown and thermal joint shall be integral; all heat transfer to the thermal radiating structure shall be accomplished through the module or subassembly mounting bolt interface. 6) All connectors used for interconnecting assemblies shall be the circular configuration and shall be so located that visible mating and inspection of all flight and test connectors are possible. 7) Assemblies shall be removable without removing thermal control elements of assemblies and without disturbing adjacent cabling. 8) All assemblies shall be of such size and weight as to be installed and removed by one man (45 pounds maximum).

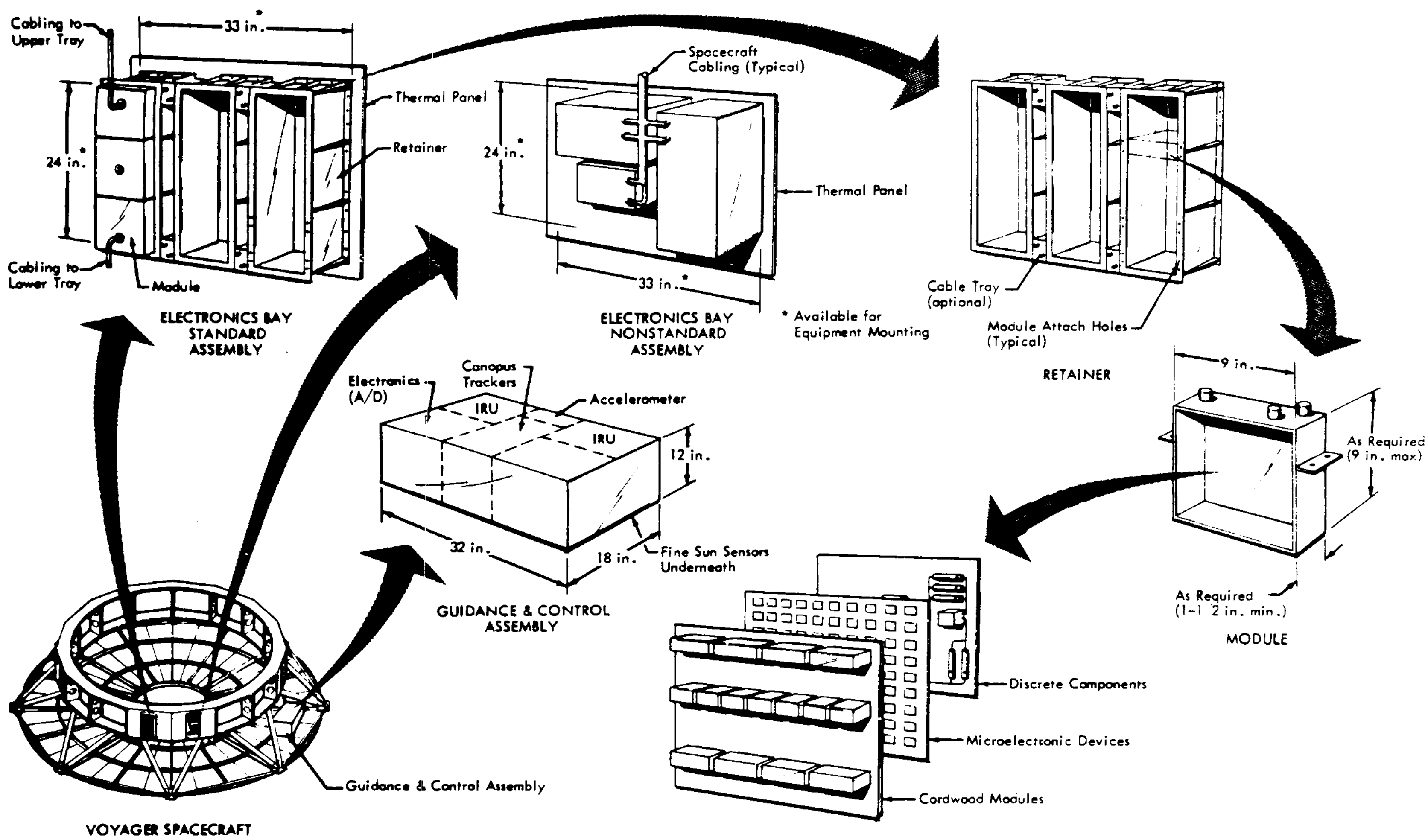


Figure 1.2.13-6: ELECTRONICS PACKAGING CONCEPT

3) Module -- The standard module as shown in Figure 1.2.13-6 has a width of 1 1/2 inches, a length of 9 inches, and a maximum height of 9 inches. The optimum height of the module shall be determined for each subsystem to accommodate the particular form factors of the components of the subsystem being packaged.

Provisions are made for bolting several modules together to form larger modules dependent on the circuitry requirements for modularization. Module covers for particular RFI requirements are provided.

The center web of the module to which printed circuit boards are laminated forms a thermal path from the electronic part on the circuit board to the thermal panel; the nominal temperature rise between the electronic part and the radiating surface of the thermal panel is expected to be less than $15^{\circ}F$. The faying surfaces of the module/thermal panel interface will be machined to a 32 microinch finish to obtain thermal contact without the use of indium foil or silicone grease.

Thermal contact can be maintained with the thermal panel by three methods depending primarily upon the amount of heat generated within each module:

a) Module contact with the bottom of the structural housing by recessing head screws through the housing baseplate and with the housing flanges bolted to the panel from the equipment side.

b) Module pressure contact directly to the panel (no bottom in the housing) with the housing flanges bolted to the panel (creating the module pressure contact). Use for low heat modules only.

c) Same as b) above except with recessed head screws through the thermal panel into the modules. This option should not be used except in severe thermal problem areas caused by high heat generated within a module. This approach increases the number of fasteners that must be removed to replace a component.

   The number of bolts or screws used to fasten the modules is dependent on the thermal characteristics of the module; 1 watt per fastener is assumed to be the nominal thermal requirement. Gapped nutplates will be used to contain any metallic chips formed in the fasteners.

The Task D packaging design for modules employs a conservative approach that is based on proven technology and processes. Cordwood modules and printed circuit boards will be used to house discrete parts; flat packs will be soldered to printed circuit boards which are laminated to the metallic center web of the subassembly housing. Z-wire or plated-through holes with a conductor soldered through the hole will be used to connect the two printed circuit sides of a printed circuit board. Multilayer boards will be used only if the mounting densities of circuit boards with circuitry on the two sides are inadequate, and if reliability increases can be achieved (proven by test and usage) by technology improvements. However, per NASA-MSFC requirements, all plated-through holes of approved multilayer boards will use a conductor soldered in place in the hole to improve reliability.

4) Interconnections -- Assembly internal wiring shall be made up as a harness and attached to the retainer and thermal panel as appropriate. The preferred module connector shall be of the circular type (NAS 1599, MIL-C-26482, and MIL-C-26500 series of connectors). However, rectangular module connectors shall be considered if particular weight and volume reduction can be achieved with no decrease in reliability.

5) Nonstandard Assemblies -- Where not practical to conform with the "standard" packaging concept as outlined in 3.2 above, the assemblies shall be bolted directly to the thermal panel.

The thermal requirements of the assemblies shall be evaluated in the layout phase of the thermal panels to effect adequate thermal control; stiffeners shall be added for mechanical support as required. Space shall be allowed for cable routing, and for stiffeners as required.

All assembly connectors shall be specified as NAS 1599 and shall be mounted on top (opposite the mounting surface) for complete visibility when being mated.

6) Guidance and Control Assembly -- The guidance and control electronics housed in an assembly 18 x 32 x 12 inches that is installed above the solar panels and outboard of the equipment and cable module as shown in Figure 1.2.13-6. The upper surface of the assembly will function as a thermal panel with louvers to control the thermal environment. The sides of the assembly excluding the thermal panel, and the sensor apertures will be covered with multilayer insulation.

The individual elements that require alignment (IRU's, accelerometer, and Canopus trackers) are first aligned to the precision machined-chassis. The chassis of the guidance and control assembly also incoroporates a prism and adjusting devices to align the assembly for correct sensor aperture viewing.

Physical Characteristics -- The packaging system elements and locations are shown in Figure 1.2.13-6.

Interface Definition -- Principal interfaces of the packaging subsystem are not identified uniquely as such, since elements of the packaging subsystem are integral with the individual electrical and electronic subsystems.

Reliability -- The assessed reliability of the packaging subsystem is included in the electronic and electrical subsystems reliability assessments.

Trade Study Summary -- Trade studies were performed to select approaches for standard assembly configurations. These trade studies are summarized in Figure 1.2.13-7.

New Technology and Development Items -- No new technology or development is required for packaging.

Growth Potential -- Of the eight thermal panels, six are used for the basic spacecraft equipment. The other panel areas are available for science and growth.

TRADE STUDY SUMMARY SHEET	SOURCE OF REQUIREMENT	TRADE STUDY NUMBER & TITLE: PACKAGING CONFIGURATION	SELECTION
		MATRIX DESIGN APPROACH	

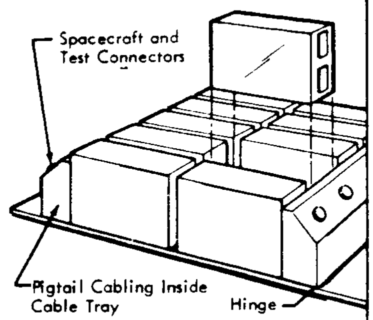
FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

Packaging configuration shall accept both standard and nonstandard assemblies for placement on a thermal panel for mechanical support, and thermal control. Modules shall be retained as a subsystem where required. All connectors shall be capable of being visible during mating. Assemblies shall be removable without disturbing other assemblies, cabling, or thermal control elements.

Trade Considerations

Simplicity
Reliability
Weight

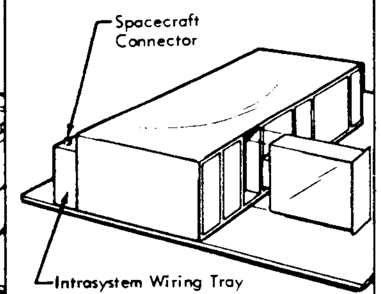
1 Hinged Cable Tray



ADVANTAGES
- Protected wiring (subsystem)
- Accessible to S/C cabling from top or side
- Integral structural-thermal joint

DISADVANTAGES
- Additional room required to open hinged wiring tray

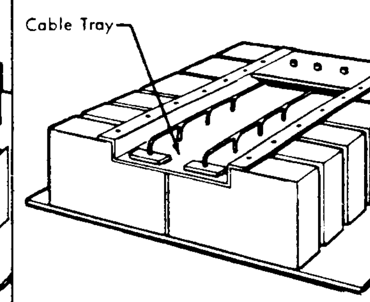
2 Lateral Plug-in



ADVANTAGES
- Integral structural thermal joint

DISADVANTAGES
- Access required on the side to remove subassembly
- Blind mating of subassembly connector

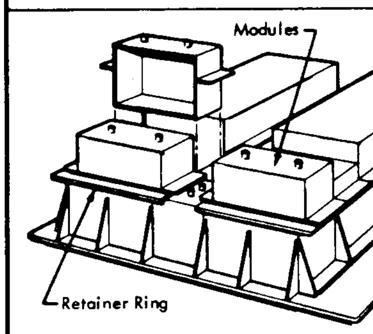
3 Exposed Cable Tray



ADVANTAGES
- Integral thermal-structural joint

DISADVANTAGES
- Wiring must be disturbed to remove a subassembly
- Possible pin limiting because of connector spatial considerations.

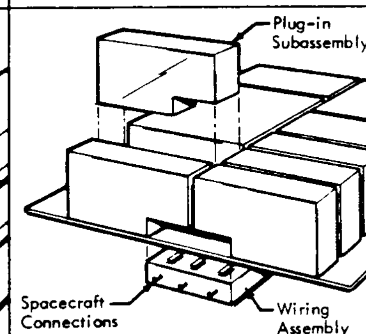
4 Module Retainer



ADVANTAGES
- Subsystem can be removed as entity
- Integral thermal-structural joint
- All cabling & connector interfaces are visible

DISADVANTAGES
- Retainer ring increases weight

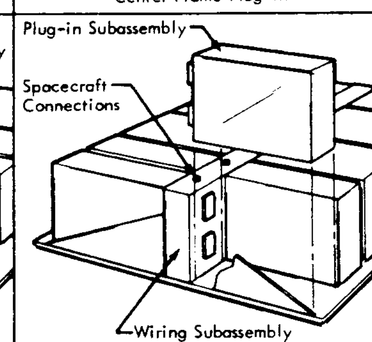
5 Vertical Plug-in



ADVANTAGES
- Short cabling runs for subsystem intraconnections
- No pigtail harnessing

DISADVANTAGES
- Blind mating of subassembly connectors
- May be pin limited due to connector space allocation

6 Center Frame Plug-in



ADVANTAGES
- Central wiring tray provides short cabling runs

DISADVANTAGES
- Requires access on sides to remove and install
- Blind mating of subassembly connectors

SELECTION

Simplicity
4, 3, 5
Reliability
4, 3, 1
Weight
5, 4, 3

SELECTED APPROACH

4

Figure 1.2.13-7: PACKAGING TRADE STUDY

2.0

MINIMUM CHANGES TO 1973 SPACECRAFT FOR 1975 - 1979 ADAPTABILITY

2.0 MINIMUM CHANGES TO THE 1973 SPACECRAFT FOR 1975 TO 1979 ADAPTABILITY

2.1 INTRODUCTION

This section describes the changes required to adapt the 1973 spacecraft to meet the 1975, 1977, and 1979 missions. The conceptual changes required are presented for the spacecraft configuration and each of the hardware subsystems. The changes required result from consideration of the following:

a) Effect of launch year on transit time, trajectories, etc.

b) Effects of science payload evolution discussed in D2-115002-4.

c) Effects of photoimaging considerations discussed in D2-115002-4.

2.2 SPACECRAFT LAYOUT AND CONFIGURATION

Changes in the spacecraft layout and configuration from the 1973 baseline result from changes in the science payload, the capsule, and the additional scientific data that must be transmitted to Earth. These changes in basic mission requirements result in changes in support structure, in solar cells to furnish the required power, and in antenna diameter to increase the communication rate. The basic changes from 1973 through 1979 are listed below:

	MISSION YEAR			
ITEM AFFECTING CONFIGURATION	1973	1975	1977	1979
Science Payload - Pounds	371	592	1,001	1,056
Capsule Weight - Pounds	6,000	7,000	7,000	7,000
Propulsion System	Designed for 1979 Mission			
Antenna Diameter - Feet	13.7	13.7	19.0	19.0
Structure	Designed for 1979 Mission			
Attitude Control	Cold Gas	Cold Gas	Hydrazine	Hydrazine
Power-Solar Panel Area - Sq Ft	290	352	414	414

2.2.1 Changes to Baseline for 1975 Mission

The increased size of the scientific payload results in a greater power requirement for the 1975 mission. This is furnished by 62 square feet of foldout solar panels. The panels are deployed after the spacecraft is separated from the shroud.

Since the structure and fuel tanks were designed for the 1979 mission, they are adequate for all the missions 1973 through 1979. Figure 2.2.1-1 shows the 1975

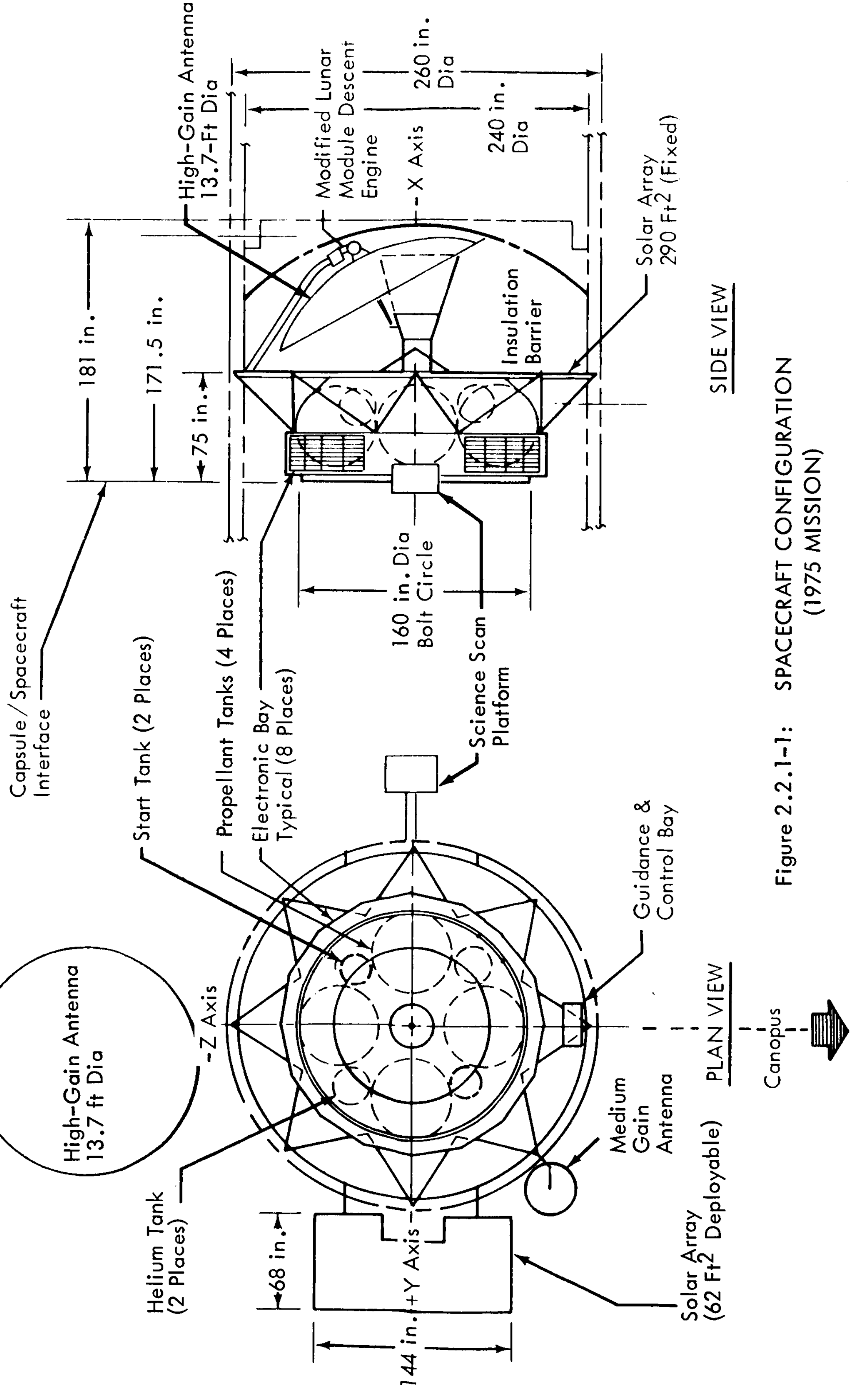


Figure 2.2.1-1: SPACECRAFT CONFIGURATION (1975 MISSION)

spacecraft with a single foldout panel. Table 2.2.1-1 is a summary weight statement comparing the 1975 vehicle with the 1973, 1977 and 1979 vehicles.

2.2.2 Changes to Baseline for 1977 and 1979 Missions

The high resolution camera requirements determine most of the changes to the baseline configuration for the 1977 and 1979 missions. The camera will be mounted between the fixed solar panels and the propellant tanks and will be attached to the propulsion support structure (see Figure 2.2.2-1). Some revision of the structure is required to accommodate the optical system. Since the camera is located off the center of gravity of the vehicle, it will be necessary to move some of the electronic bays to static-balance the vehicle. Likewise, it will be necessary to move one of the two helium bottles to make room for the camera. Since the structure and fuel tanks were designed for the 1979 mission, they are adequate.

The data requirements of the large optical system necessitate the addition of 124 square feet of solar panels to the baseline configuration. A large (100 watt) output traveling wave tube will be used for communication requiring additional changes in the communication bay. Also, the 13.7-foot diameter high gain antenna will be replaced by a 19-foot diameter antenna to improve communication capability. The larger antenna will require additional pointing accuracy.

The large optical system will also require more precise stabilization and control than the smaller systems on 1973 and 1975 missions. The camera will be body-mounted, necessitating maneuvering the entire spacecraft to point the camera. This maneuvering will require additional reaction control capability.

2.3 HARDWARE SUBSYSTEMS

This section describes the changes required to adapt each of the hardware subsystems to meet the 1975, 1977, and 1979 missions.

2.3.1 Power Subsystem

Increases in the power requirements of the science subsystem for the 1975, 1977, and 1979 missions and the addition of a 400-watt TWTA load for the 1977 and 1979 missions are major factors affecting power subsystem adaptability to these missions. Subsystem components that will be significantly affected by these load increases include the batteries, regulated inverters, and the solar array.

During the occulted portion of Mars orbit of the 1977 and 1979 missions, the science payload and the higher-powered TWTA require more battery energy than is provided for the 1973 mission. Tabulated below are the 1973-1979 mission science and TWTA power requirements during the 90-minute occulted portion of the Mars orbit.

Science Power		TWTA Power
1973:	45 watts	200 watts
1975:	45 watts	200 watts
1977:	90 watts	400 watts
1979:	82 watts	400 watts

Table 2.2.1-1: SUMMARY WEIGHT STATEMENTS 1973 THROUGH 1979 MISSIONS

	WEIGHT (Pounds)							
	1973 Mission		1975 Mission		1977 Mission		1979 Mission	
Structure		539		539		539		539
Capsule Support Truss	129		129		129		129	
Star Truss	318		318		318		318	
Special Purpose Fittings	92		92		92		92	
Propulsion		1,799		1,799		1,799		1,799
Lunar Module Descent Engines	409		409		409		409	
Propellant System	895		895		895		895	
Pressurization System	471		471		471		471	
Engine Support & Misc.	24		24		24		24	
Equipment and Instrumentation		2,595		2,934		3,310		3,365
Structure (Equip. Compt. & Thermal)	189		189		222		222	
Guidance and Control	139		139		139		139	
Instrumentation	545		545		549		549	
Electric Power	573		635		797		797	
Electric Networks	324		324		324		324	
Temperature Control	184		184		184		184	
Attitude Control System	124 [1]		171 [1]		54 [2]		54 [2]	
Science Equipment	146		155		40		40	
Science Payload	371		592		1001		1056	
Contingency		257		258		277		282
Total Dry Spacecraft		5,190		5,530		5,925		5,985
Residuals		379		379		382		382
Total Inert Spacecraft		5,569		5,909		6,307		6,367
Usable Propellant		10,951		12,171		12,873		12,873
Fuel	4,190		4,650		4,860		4,860	
Oxidizer	6,700		7,430		7,770		7,770	
Attitude Control Fluid	61 [1]		91 [1]		243 [2]		243 [2]	
Total Spacecraft @ Liftoff		16,520		18,080		19,180		19,240
Capsule		6,000		7,000		7,000		7,000
Total Planetary Vehicle @ Liftoff		22,520		25,080		26,180		26,240
Adapter Provisions		240		240		240		240
Total Planetary Vehicle Plus Adapter		22,760		25,320		26,420		26,480

[1] N_2 System

[2] N_2H_4 System

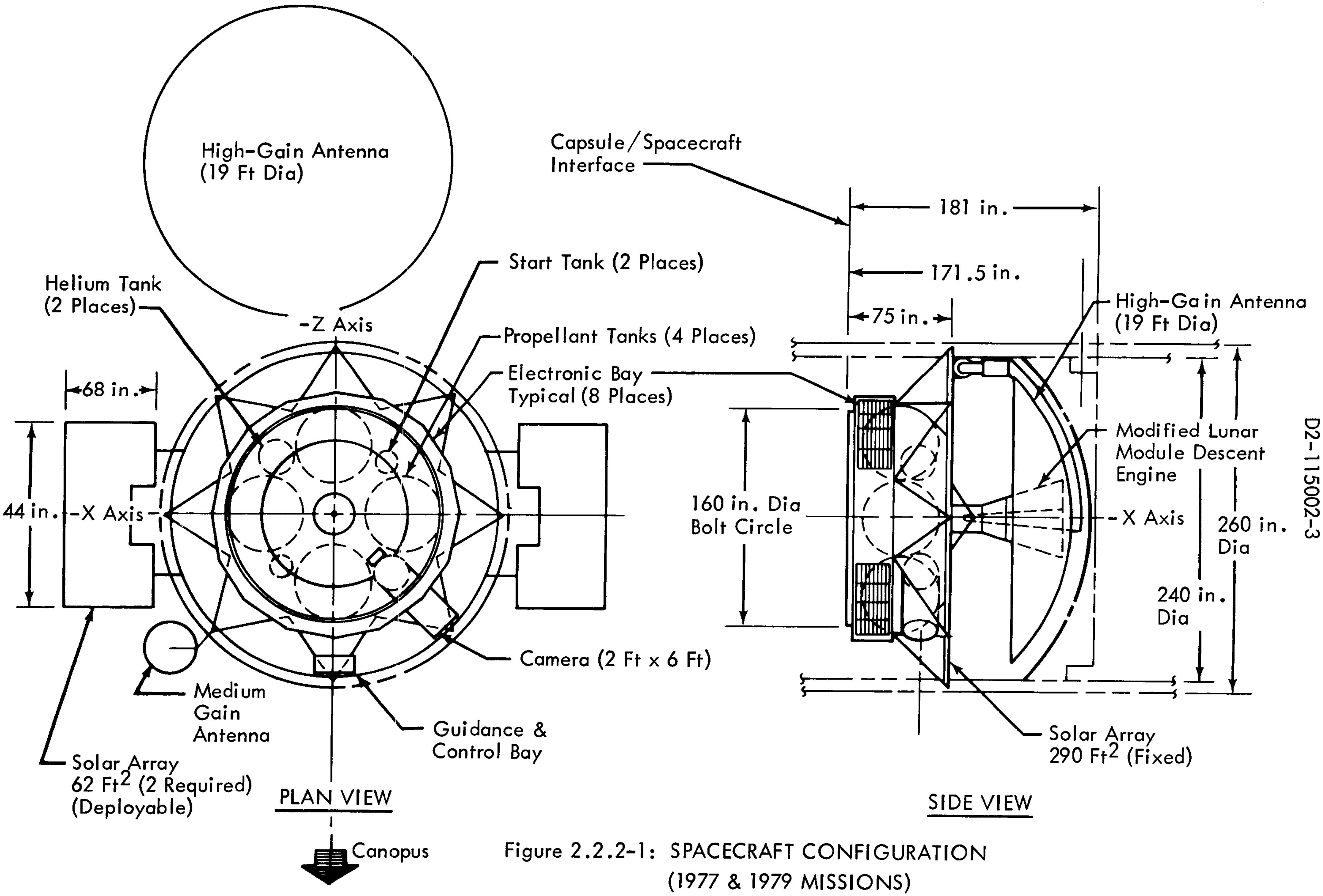


Figure 2.2.2-1: SPACECRAFT CONFIGURATION
(1977 & 1979 MISSIONS)

Paralleling a fourth battery and battery charger for the 1977 and 1979 missions with the three batteries selected for the 1973 mission satisfies the increased energy storage requirements. Table 2.3.1-1 illustrates battery sizing for the 1973-1979 missions.

Table 2.3.1-1: BATTERY SIZING FOR 1973 - 1979 MISSIONS — Critical Flight Regime: Orbital Occultation

	Mission			
	1973	1975	1977	1979
Power Required (watt hours)	940	940	1336	1320
Battery Rating (watt hours)	2733 [1]	2733 [1]	3645 [3]	3645 [3]
Depth of Discharge (percent)	52 [2]	52 [2]	49 [4]	48 [4]

[1] Based on 3 batteries with 38 series-connected, 20 A-H cells operating at 1.2 volts/cell.
[2] Based on 2 of 3 batteries operating.
[3] Based on 4 batteries with 38 series-connected, 20 A-H cells operating at 1.2 volts/cell.
[4] Based on 3 of 4 batteries operating.

The science payload peak power and additional TWTA power requirements during the illuminated portion of the Mars orbit affect sizing of the solar array and regulated inverters for 1975-1979 missions. Tabulated below are the peak science and TWTA loads for the 1973-1979 missions.

Peak Science Power		TWTA Power
1973:	163 watts	200 watts
1975:	260 watts	200 watts
1977:	360 watts	400 watts
1979:	360 watts	400 watts

These increased loads, plus the increased power required to charge a fourth battery for the 1977 and 1979 missions, and increased regulated inverter losses based on 75% overall inverter efficiency, will require the use of deployable solar panels in addition to the 290 square feet of fixed solar panels required for the 1973 mission. Tabulated below are the minimum array area requirements for the 1975-1979 missions based on these increased power requirements and based on the solar array degradation and design factors used for calculating the 1973 mission requirements:

1975: 307 square feet

1977: 414 square feet

1979: 414 square feet

The 1973 mission science loads are supplied a.c. power from three 230-watt regulated inverters. The science loads predicted for missions subsequent to 1973 will require at least one additional inverter of the same rating or three inverters of higher rating.

The number of batteries, the solar array area, and the inverter ratings required for the 1975-1979 missions are summarized in Table 2.3.1-2.

Table 2.3.1-2: MINIMUM CHANGES TO 1973 SPACECRAFT POWER SUBSYSTEM FOR 1975 - 1979 ADAPTABILITY

	Mission			
	1973	1975	1977	1979
Batteries (Number)	3	3	4	4
Solar Array Area (Square Feet)	290	352	414	414
230-Watt Inverters (Number)	3	4	4	4
or				
280-Watt Inverters (Number)	-	3	4	4
or				
330-Watt Inverters (Number)	-	-	3	3

2.3.2 Guidance and Control Subsystem

The changes required for the 1975, 1977, and 1979 missions are a result of the following:

1) The mission time is increased for both transit and orbit.

2) The larger capsule, deployed solar panels and larger spacecraft science payload increase the moment of inertia in both transit and orbital operations.

3) The high resolution science package requires maneuvering the spacecraft to point the camera.

4) The larger antenna and high resolution camera in 1977 and 1979 required better spacecraft pointing accuracy.

The implication of items 1) and 2) is a modest increase in reaction control propellant. Thruster sizes selected for Task D appear adequate. However, item 3) implies a very large increase in maneuver impulse. This increase is required to perform a three-axis photo maneuver and return to celestial reference for each photo frame taken and assumes 360 photo maneuvers.

To accommodate this increase will require use of a more weight-effective means of maneuvering. One attractive approach is to adopt a hydrazine rocket reaction control for maneuvers. The limit cycle reaction control could be cold gas nitrogen or gas from hydrazine decomposed by a catalytic reactor and stored in a plenum chamber.

A plenum/rocket hydrazine hybrid reaction control system would weigh 300 pounds for a photo mission in 1977 and 1979 as opposed to 1238 pounds for an all cold nitrogen gas system.

An alternative approach to the 1977 and 1979 mission requirements would be to complement the reaction control system with momentum exchange devices for attitude control. The system would be sized to perform the necessary attitude maneuvers for the camera experiments. A system sized in such a manner would also provide the required pointing accuracy and stability for a laser communications experiment and would adequately cope with gravity gradient, aerodynamics and solar pressure disturbances during the process. Current data for a control moment gyro (CMG) system for these purposes shows a weight of 132 pounds for three-axis control with a spin power requirement of 72 watts. The impulse requirement to counteract spin-up torques would be 12 lb-sec. Additionally such a control moment gyro system could provide backup to the inertial reference unit. Should operation be required during solar occultation, it may be possible to discontinue spin power to the CMGs and maintain control in a spin-down mode, thus avoiding additional battery weight penalties.

2.3.3 Data Storage Subsystem

The data storage subsystem design for the 1973 mission profile provides a total bulk storage of 1.2×10^9 bits. Assuming this design to be modified to include previous DAE functions such as formatting, multiplexing, and buffer storage for the science payload, the bulk storage capability utilizing eight tape recorders provides sufficient flexibility to match the nominal transmission capability of 4×10^{-11} bits of science data for the 1975 mission profile. This capability assumes a 100% utilization of the orbital period for transmission.

The proposed 1977 and 1979 mission profiles indicate a revision of the science imaging experiments to include a photographic subsystem with film storage. This evolution of the science payload would result in reducing the overall bulk storage requirement. Therefore, only recorders to satisfy engineering, capsule/lander, and auxiliary science sensor data requirements would be required. It appears that the 1973 baseline would be adaptable and capable of meeting foreseen 1977 and 1979 mission data requirements.

2.3.4 Telemetry Subsystem

The telemetry subsystem design for the 1973 mission profile considered and incorporated the data modes necessary to cope with the 7-db variation in performance due to Mars-Earth range profile. In addition, the design permits a large degree of flexibility in adjusting to a wide variety of input data requirements, subcarrier baseband composition, and subcarrier modulation techniques. As a result there is no major design change anticipated to satisfy 1975 mission requirements.

However, for the 1977 and 1979 mission profiles, it is anticipated, as a result of the photoimaging data requirements, that modulation and baseband composition will be altered. As a result of high-data-rate trade studies (see D2-115002-2, Section 3.9), the upper subcarrier channel would be modulated by an analog baseband utilizing either a direct frequency modulation or a vestigial sideband amplitude modulation technique as utilized in Lunar Orbiter. Either of these modulation techniques would be compatible with a lower subcarrier channel and with the phase-modulated carrier as currently postulated.

These analog modulation techniques could provide under nominal performance parameters (ERP = 88.3 dbm) an analog transmission rate of 27 kHz. This transmission bandwidth would correspond to a total of 2×10^{13} information bits during a 24-month interval. A corresponding digital link using coded PSK modulation would have a data rate of 320,000 bps at an ERP of 90.6 dbm.

2.3.5 Radio Subsystem

The radio subsystem design is adaptable to all 1975 and 1979 mission profiles as shown by the performance capabilities of the command, range tracking, and carrier tracking channels. However, as a result of the science payload evolution for 1977 and 1979 missions, the overall effective radiated power requirements increase from 79 dbm to 88.3 dbm. This requirement would result in increasing the output power of the power amplifier from a nominal 65 watts to 140 watts. The choice of power amplifier type can not be made at this time due to the development status of 100- to 200-watt power amplifiers.

2.3.6 Antenna Subsystem

The antenna subsystem design satisfies 1975 mission requirements in all areas of spacecraft transmission and reception with the possible exception of the capsule relay receiving antennas. Depending upon the capsule/lander mission profile relative to the spacecraft's orbital parameters, a more extensive antenna system may be required to satisfy the wide range of look angles and increased antenna gain requirements anticipated for the subsequent missions.

As was discussed in Section 2.3.5, the increased data transmission rate for the 1977 and 1979 missions requires an ERP of 88.3 dbm corresponding to a 140-watt power amplifier and a high gain antenna diameter of 19 feet. To achieve the necessary antenna gain corresponding to an ERP of 88.3 dbm, the antenna pointing error per axis must be reduced from ± 0.6 degree to ± 0.35 degree. This pointing requirement may necessitate an active control or tracking loop and/or increased spacecraft stabilization requirements.

In view of the above requirements for a wideband RF link, a laser experiment is recommended for the 1975 mission to establish feasibility of a wide-band laser link with arc-second pointing capability. Depending upon the results of this experiment and corresponding hardware development status, a laser link might be incorporated in the 1977 and 1979 missions relaxing the RF link requirements in the areas of antenna diameter, antenna pointing accuracy, and transmitter output power.

2.3.7 Computing and Sequencing Subsystem

No changes are required to adapt the computing and sequencing subsystem to meet the 1975, 1977, and 1979 missions because the C&S subsystem with its computer-like organization is not mission-dependent. Variances in mission objectives and mission profile can be accommodated without hardware changes. (The affect of including the DAE function is discussed in Section 1.2.8.8).

2.3.8 Structural and Mechanical Subsystem

Structural -- No basic changes are required to the 1973 spacecraft structure to accomplish the 1975 and 1979 missions. The 1973 spacecraft structure is specifically designed to carry a 7000-pound flight capsule and a propulsion system with a usable propellant capacity of 12,873 pounds.

Mechanical -- Spacecraft mechanisms are designed specifically for the 1973 mission. Where design requirements for mechanical systems proposed for use on subsequent missions can be reliably established during the 1973 mission design phase, their incorporation into 1973 mechanical systems will be considered.

2.3.9 Pyrotechnic Subsystem

No changes in design concept are required to adapt the pyrotechnic subsystem to meet the 1975, 1977, and 1979 missions. Additional drivers and EED's will be added as required by changes to the other spacecraft subsystems.

2.3.10 Temperature Control Subsystem

The changes in the baseline 1973 Voyager spacecraft thermal control subsystem required to meet the 1975 to 1979 missions are detailed in the following paragraphs. These major changes are (1) capsule RTG radiator, (2) high heat load TWTA, (3) fixed-camera package, (4) increased science heat load, (5) hydrazine attitude control system, and (6) deployable solar panels.

2.3.10.1 Capsule RTG Radiator

The addition of a capsule RTG radiator to the lower capsule sterilization canister will result in an increase in the solar array temperatures for those areas extending beyond the projected spacecraft bus envelope. The RTG radiator will also result in a reduction in the radiator louver assembly performance. For example, a 533°K (500°F) capsule RTG radiator adjacent to a radiator louver assembly will reduce the performance by 36% at the upper design limit of 297°K (75°F). The capsule RTG radiator size, shape, and location directly affect the spacecraft heat rejection capability. A preferred capsule RTG radiator location would be between the equipment bays. A 533°K RTG radiator will also result in an additional solar array heat load of about 49 watts per ft^2 over local areas under the radiator.

2.3.10.2 **TWTA Heat Load**

The increase in the radio subsystem power input requirements from a 200-watt TWTA to a 400-watt system represents a significant increase in the total spacecraft heat load. Depending upon the area over which this additional heat load is generated, an active temperature control system may be required. Assuming a 25% exchange TWTA, the minimum area required to dissipate 300 watts of thermal energy, assuming a uniform distribution of the heat load, at 322°K (120°F), is 7 square feet and, at a temperature of 305°K (90°F), the area required is 8.7 square feet. The baseline Voyager configuration radiator area available is only 6 square feet per radiator plate.

2.3.10.3 Fixed-Mount Camera

The addition of a fixed-mounted camera package in the spacecraft requires that the spacecraft be maneuvered on each orbit (off-Sun maneuver) for about 30 minutes to take photographs of the Martian surface. Generally, the spacecraft is designed to withstand a 60-minute off-Sun maneuver near Earth; therefore, the 30-minute maneuver is not considered serious. However, the camera package is a hermetically sealed unit requiring close external thermal control and close temperature and humidity control inside the camera. Special thermal control methods are required to prevent "fogging" of the lens because of the locally condensed moisture inside the camera. Such a thermal control method may be a thermal door similar to that used on the Lunar Orbiter. Additional lens heaters will be necessary to hold the light tolerance to avoid defocusing problems.

2.3.10.4 Science Subsystem

The increase in the science subsystem heat load will require a larger science scan platform radiator-louver assembly. The science subsystem heat load increases 195 watts (from 165 to 360 watts) of which 55 watts are body-mounted science equipment and 140 watts are for the camera. The baseline heat load is 120 watts at the science scan platform and requires a radiator of approximately 1.46 square feet. The radiator size will increase to about 4.75 square feet.

2.3.10.5 Attitude Control System

The proposed use of hydrazine as an attitude control system working fluid results in a propulsion module temperature control range of 297 to 317°K (75 to 110°F) instead of the present 278 to 317°K (40 to 110°F). The propellant lines and thrusters will require insulation with a conductance of less than 10^{-3} watt/ft^2-°R.

2.3.10.6 Deployable Solar Panels

The additions of deployable solar panels to the baseline spacecraft results in a degradation of the radiator-louver assembly performance because of the reduced view of space. Also, the additional heat flux from the deployed solar array may cause the radiator-louver assembly performance to become unstable. To correct this instability, the louver travel would be restricted to a value less than 90 degrees.

ERRATA SHEET

Page	Errata
1-1	In fifth paragraph, first line, change "requirement" to "requirements".
1-61 and 1-62	In upper part of plan view change "+Z AXIS" to "-Z AXIS".
	Below +Y Axis in plan view change "(TYP. 4PL)" to "(TYP. 2 PL)" for propulsion louver assembly callout.
1-92	In third line of paragraph 3), change "parrallel" to "parallel".
1-149	In paragraph 3), line 5, change "Table 1.2.3-1" to "Table 1.2.3-3".
1-195	In third paragraph, third line delete "(Note 1)".
1-231	In 1.2.8.3, change weight to 69 pounds.
2-8	In first paragraph of 2.3.3, change "4×10^{-11}" to "4×10^{11}".
2-9	In last line of 2.3.4, change "90.6 dbm" to "92.4 dbm".